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OBJECTIVES OF CONFERENCE:

To present technical contributions summarizing the status of current research in expandable structures. . . . .

To encourage development of new research in expandable structures. . .

To provide a forum of authorities to critique current research, and propose new research and applications for expandable structures. . . . .

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ACKNOWLEDGMENT: The Technical Committee expresses their sincere appreciation to Mr. [illegible], Vice President, Research and Engineering, National Cash Register, for his stimulating introductory remarks.

TABLE OF CONTENTS

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BY

MAJOR GENERAL MARVIN C. DEMLER

TO

THE CONFERENCE ON AEROSPACE EXPANDABLE STRUCTURES

DAYTON, OHIO

22, 23, 24 OCTOBER 1963

Thank you, Colonel Hawkins. It is indeed a pleasure to speak to you today, the first true conference devoted exclusively to Expandable Structures. However, as you can see from our Agenda for the next three days, Expandable Structures are anything but exclusive, being quite all-inclusive in their application. That there is a broad spectrum of application is evidenced by the fact that four of our seven Research and Technology Laboratories are engaged in various phases of research and development in this area. The Propulsion and Flight Dynamics Laboratories, which are co-sponsoring this symposium in cooperation with National Cash Register as well as The Avionics Laboratory have been and are devoting extensive effort to specific applications, while the Materials Laboratory is providing the basic know-how in expansion, rigidizing, and fabrication techniques.

Expandable Structures have been with us for quite a long time, so long perhaps that some people may wonder why we need to hold a symposium on the subject. Anyone who has used an umbrella, a bellows camera, a parachute or a pneumatic life jacket has been exposed to expandable structures of an everyday variety. It has been the advent of airborne and space operations, however, which has brought a sense of urgency and discrete focus on the subject, and which justifies a symposium such as this one. The national importance of this meeting is very apparent because not only are there Air Force papers and session chairmen, but also session chairmen and papers from the Army, NASA, industries and universities.

The principal impetus for accelerated efforts on expandables is related directly to the cost and penalties of transporting large, bulky, and heavy structures around and about the earth as well as into space. Although aircraft, aerospace vehicles, and boosters are constantly being increased in payload capacity, it is still impossible to transport large volume structures in one piece. Consequently, these structures must be assembled at their final destination, resulting in a great loss in mission efficiency.

Expandable structures afford the structural designer the opportunity of sending to the site a one piece packaged structure which can be erected and made operational in a minimum of time. Because of this, the resulting Research and Development programs have been pursued by practically every governmental and industrial agency having a research and development responsibility or capability, and a tremendous variety of applications and state-of-the-art breakthroughs have resulted.

In May 1961, The House of Representatives' Committee on Science and Astronautics conducted a hearing on inflatable structures in space and because of the evident broad spectrum of applications, issued a request that, and I quote "The Committee would like assurance from the National Aeronautics and Space Administration and the Department of Defense that coordination of work in Expandable Structures is not being overlooked simply because the programs are so varied and so scattered". I believe that, in a very real sense, this symposium today demonstrates that such coordination is being effected, and that views and ideas are being freely exchanged.

So, in getting down to the purpose of the conference, the principal objectives may be stated as being:

To present technical contributions summarizing the status of current Research and Development in Expandable Structures....

To encourage the undertaking of new research in expandable structures:-

To discuss potential applications and

To provide a forum of authorities to critique current research, and propose new research and applications for expandable structures........

The term 'Expandable Structures' has unusually broad connotations, as you all know. It encompasses, in its most general meaning, any structures which change their geometry by expanding or unfolding to permit the performance of a task. Generally speaking they can be classified by their method of expansion and stabilization into three classes:

Inflated, pressure-stabilized structures;

Inflated, rigidized structures as by the use of foams or coatings which become rigid and

Mechanically expanded, framework stabilized structures.

All of these types will be discussed during this symposium. As far as the Air Force is concerned, the application of expandable structures falls into four general categories.

First and probably the most simple are the popularly known large shapes which are expanded in place on earth or in space and remain there. These include items such as:

Shelters or space "stations";

Life raft vehicles;

Decoy devices, and

Communications and reflecting satellites

A most promising investigation that fits this category is presently being conducted by the Air Force Flight Dynamics Laboratory and the Materials Laboratory. They are studying a telescoping configuration that could be used for a space station. It would have the advantage of multi-wall construction with a compact configuration for launch. As an expanded structure, it would have predictable structural integrity that would contribute to reliability and safety in space operations.

The second category would be shapes which are expanded in space or upper atmosphere and later re-enter. These might include:

Mission vehicles;

Emergency escape vehicles;

Rigid vehicle configurations which are expanded to achieve variable geometry;

Decelerators, and

Booster and crew recovery devices.

In this category falls such devices as the Rogallo kite and the inflatable paraglider which the Air Force Flight Dynamics Laboratory and the Materials Laboratory are investigating as a concept for an emergency escape vehicle. It is envisaged that this emergency paraglider will have a rigid boom for a central keel that will house the crew and be inflated for flight after separation by internally stored high pressure gas.

The third category would then include the smaller shapes which would be components of mission vehicles. These might include:

Camera or telescope structures;

Antennae, collectors, and mirrors;

Bumpers or shock absorbing devices, and

Couplings, as between vehicles or components

An example of one of the Air Force's advanced development programs which includes expandable structures work in this third category is the Advanced Solar Turbo-Electric Concept. It is generally; referred to by the acronym ASTEC, and is managed by the Air Force Aero Propulsion Laboratory. This program will place into orbit a forty-four and one-half foot diameter Solar Energy Collector with dynamic energy conversion equipment. This unit will be designed to produce fifteen to twenty-five kilowatts of electric power on a continuous basis with at least a one year operating life. The Solar Collector will be an expandable rigidized structure weighing about two tenths of a pound per square foot of area. This is much less than any conventional rigid metal collector, and provides a much more compact launch configuration.

The final category of applications is that of ground-based facilities or devices, primarily shelters and antennae. The Propulsion Laboratory project on functional shelters for aerospace and limited war falls in this category. This program will give the Air Force a capability for rapidly and efficiently providing personnel and systems shelters at limited war sites, aerospace sites and in the future, extraterrestrial sites. Concepts which are being investigated for these purposes include air supported, foamed-in-place, and expandable core structures. We have to date successfully rigidized shelters of seven feet in diameter under terrestrial and spatial conditions. This next summer a twenty-six

by thirty foot expandable shelter will be rigidized under simulated limited war conditions.

As we proceed with this symposium through the next few days, many more examples of expandable structures application will be illustrated which will fall into one or more of the categories I have mentioned and without a doubt will give us an insight into the solution of many problems.

However, a word of caution. We must be careful not to get carried away with an expandable structure approach to every problem. It should not be a purpose of this conference to indiscriminately offer an expandable structure solution to all future aerospace structure and logistic problems. I believe that this conference should and will indicate that the expandable structure is one of the very useful tools available to aerospace system developers.

Technology in the expandable structures area has been growing at a rapid rate and designers have recognized many potential applications. Because of physical, mechanical and propulsive limitations, expandable structures can alleviate many of our transportation and logistic problems in terrestrial and space operations.

Development effort within the Air Force on the different categories of application is carried on under several program elements. The basic inflation and rigidizing techniques fall quite naturally into our Exploratory Development, or what used to be called Applied Research, Program. Discrete component applications also tend to be limited to the Exploratory Development Program level of activity, but sometimes find their way into the system and advanced development efforts under the sponsorship of the primary systems or subsystem which requires the expandable component. The ASTEC Solar Collector is an example of this.

Applications where expandables are the primary structure can exist in any of several development levels, from Exploratory Development, through Advanced Development, to Systems Development, depending on the magnitude of effort necessary to satisfy requirement. A space station, for example, would require systems type resources support, but such items as shelters, antennae, and parachutes, which involve relatively small development effort, can remain at the exploratory level until required for specific application. We tend towards keeping expandable structure research and development effort at the Exploratory Development level until a discrete application requirement is apparent, and then incorporate the expandable solution into the development program of the specific application.

In conclusion, I would like to leave you with a little story about the executive of a firm that may be represented here today. He was extremely concerned about the company's future, particularly as it related to expandable structures work which was a sizeable part of the corporation's business, and things weren't going too well. Now, being not only concerned, but also being a somewhat superstitious fellow, this executive decided to utilize the services of a traveling clairvoyant who happened to be operating nearby. He sought out this fortune teller, and proceeded to pour out his soul to this bearded and turbaned soothsayer. Upon receiving appropriate remuneration for his services in advance, the fakir gazed long and mysteriously into his crystal ball. After a seemingly endless period of time, he raised his

head and asked, "Did you say your products sometimes depend on inflationary techniques?" "Yes, yes", was the eager reply. The fortune teller went on, "And did you not also say that these products were expandable?" "Of course!" the executive replied impatiently. "Well," exclaimed the prophet triumphantly, "you have nothing to worry about. I see this day a tax bill being debated in the Congress. There are heated views being expressed by opposing sides. But, no matter, you cannot lose, for one side says it is bound to lead to inflation, while the other maintains that expansion will be the end result. There should be nothing but success for your company."

Obviously (ladies and) gentlemen, this anecdote reflects a gross misunderstanding of our subject matter for today, which is in fact quite often the case. However, I hope that this symposium by focusing the spotlight on discrete applications and techniques, will help show the way to the effective and efficient utilization of expandable techniques.

THE SHAPE OF THINGS TO COME
IN EXPANDABLE SPACECRAFT

By D. G. Younger

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ABSTRACT

The interdependence of structural concepts, materials, and the human factor in the design of expandable spacecraft are reviewed. A wide range of non-rigid and semi-rigid concepts, either under development or recently conceived, are discussed with emphasis on mission requirements and booster capability. The merits of the related composite wall cross sections for these configurations are considered in detail. Specific topics covered are those of the geometrical aspects of packaging and deploying of expandable concepts, meteoroid penetration resistance, thermal control, attenuation of penetrating radiation, and structural efficiency.

THE SHAPE OF THINGS TO COME IN EXPANDABLE SPACECRAFT

By D. G. Younger

INTRODUCTION

In the specialized field of spacecraft structures there is no area of design more exciting than that of expandable structures. It is a new area that has already demonstrated its ability to trigger the imagination and challenge the impossible. At this early date, we find that visions of erecting large satellite islands or space stations have become commonplace, and new concepts and approaches are now being introduced more rapidly than they can be evaluated. In fact, for many missions in space, technical illustrators and artists have moved far ahead of engineering, and there is little doubt that these people will have a strong influence on the shape of things to come in expandable concepts. Nevertheless, we must not lose sight of the fact that both the feasibility and the merits of any such clever concept, during this decade, strongly depend on the present state of the art and the predictable technological developments that will accrue over this period of time. The justification of employing expandable structures in space will depend heavily on our ability to establish meaningful guide lines for development, the flexibility we exercise in applying new knowledge, and the attention we give to areas where the inadequacy of technology will require intensive and continuous research.

The primary advantage gained by the use of expandable structures stems from the fact that large structures or elements of structure can be collapsed and packaged into compact volumes for the boost phase. The compact volume is desirable not only from the point of view of the aerodynamics and control of the launch vehicle, but also with respect to the performance of the critical procedures of abort of manned-craft during boost.

Once the boost phase has been completed and the packaged structure arrives in orbit, the volume or shape of the structure can be increased or changed by any of a number of schemes of unfolding, telescoping, or purely inflating of components. Such structures are now being developed for unmanned space application in the form of reflectors, solar collectors, scientific equipment packages, and re-entry structures. Applications to manned spacecraft, however, are an order of magnitude more complex and, therefore, are chronologically further down stream. Nevertheless, many government and industrial laboratories have been looking far ahead and, as a result, have become deeply involved in the many materials and structural problems attendant to the more promising concepts. These include lunar housing, re-entry vehicles, zero-gravity (non-rotating) spacecraft, and the extremely large, rotating, space-stations.

For the manned operational mission, the expandable space structure has a number of requirements imposed on it which changes its character considerably from that of merely a shape-changing and load-carrying configuration. That is, the manned expandable modules must be compatible with the constraints imposed by the human requirements relating to internal thermal control, shielding against meteoroids and penetrating radiation, hermetic integrity, and, in the case of re-entry vehicles, the isolation of man from structural heating.

In the following sections of this general review, the problems attendant to the manned expandable spacecraft will receive emphasis; however, in many instances results or conclusions will be found applicable to the unmanned structures as well. Topics to be discussed will include payload capabilities, structural configurations, the environment, and materials selection. The structural configurations presented include those basically classified as inflatable (flexible) structures, unfurling (or unfolding) panel structures, and rigid-module, deployable structures.

MISSION PARAMETERS AND PAYLOAD CAPABILITIES

MISSION REQUIREMENTS

The structural design of a spacecraft, whether expandable or rigid, must begin with a specification of the mission, crew size, requirements for logistic support, and a reasonable understanding of the induced and natural environments to which the structure will be subjected. Missions of prolonged duration, for example, would obviously introduce many problems relating to cumulative damage of materials and components as well as human factor problems associated with weightlessness, tissue-damaging radiation, meteoroid penetration, and possibly the need for ecological systems and/or logistic support. All of these considerations could be expected to drastically influence the complexity and, therefore, the related merits of a structural concept.

The number of crew members and their assigned duties will greatly influence the structural requirements. The degree to which man will be required to maintain and service (internally and externally) the spacecraft and to initiate measures for his survival will directly affect the reliability criterion employed in configuration development. In addition, mission requirements calling for rendezvous with other spacecraft for purposes of crew exchange or re-supply will introduce structural complexities necessitated by the mechanical problems of docking, latching, and sealing, and the dynamic problems associated with docking impulses.

A large space station in a fixed orbit would be ideal as an astronaut training and rehabilitation center, observation post, or supply depot. Such a craft would probably be a gravity-induced vehicle of large diameter (say 100 foot diameter), and due to its large size, would be constructed as either an expandable concept or as an erected-in-space concept (or possibly as a combination of the two). For large structures such as this, the merits

of the expandable concept against that of the erected-in-space concept would depend strongly on the mission definition and the space logistics to be facilitated. During the past several years, Goodyear Aircraft Company, North American Aviation, Inc., Lockheed Aircraft Corp., and others, have studied the space station concept very carefully, but, as yet, it is not clear to what extent a variety of missions will affect the merits of any one structural concept. Some of the typical space missions that may be ascribed in the next decade to either the space station or the zero-gravity vehicle have been summarized in Table 1.

TABLE 1

TYPICAL MISSIONS FOR EXPANDABLE SPACECRAFT

SCIENTIFIC LABORATORY

- Scientific Research
  - Space Medicine
  - Bio-Astronautics
  - Space Physics
- Technological Development
  - Materials and Structure
  - Components and Subsystem
  - Shielding Techniques
- Launching Scientific Probes

SPACE TERMINAL

- Lunar and Planetary Missions Initiated and Terminated (Manned and Unmanned)
- Supply Depot - Moon Base
- Refueling Station

CONSTRUCTION DEPOT

- Assembly and/or Expansion of Existing Lab
- Assembly and Repair of Spacecraft

COMMERCIAL APPLICATIONS

- Communications Center
- Meteorological Station
- Navigation Aids

MILITARY STATION

- Reconnaissance
- Observation Post
- Command and Control Center
- Surveillance

TRAINING STATION

- Conditioning and Rehabilitation of Space Crews

EXPANDABLE PAYLOADS AND BOOSTER CAPABILITY

During the next several years, booster performance and the related payload capability will limit the size and weight of the payload packages that may be propelled into orbit. Furthermore, during the next few years when such payload capabilities are lowest, the uncertainties in the space environment and its effects will dictate overweight (conservative) designs. Thus, ambitious programs utilizing large spacecraft, new materials, and new

design can be expected to attract favor rather slowly. The immediate trend in expandable structures will be to utilize the proved methods of fabrication and the predictable response afforded by rigid module construction. Furthermore, it will be important to reduce, wherever possible, the complexity of the mechanics for expanding the structures. That is, the semi-rigid expandable structures which utilize telescoping schemes or simple rotation of rigid modules can be expected to be the earliest of the ⅼ ın-proved expandable spacecraft.

Since booster capability plays such an important role in the evolution of expandable manned spacecraft, information has been gathered or weights of payloads that can be placed into a near-earth orbit using the booster systems contemplated during the next several years. This information is shown in Figure 1. Superimposed on this Figure are several unmanned expandable structures representative of present day applications and a number of the more sophisticated concepts that show future promise as manned vehicles or booster recovery systems. These vehicles will be discussed in more detail in a later section.

Briefly, the structures depicted on this graph are as follows. The lower-most is that of the Echo series of Satelloons which are now famous for being the first of the inflatables to arrive in space. Next, is a sketch indicating the Ranger vehicle which is representative of the numerous scientific experiments that use expandable construction in the form of unfolding panels for deploying solar-cell arrays into their most efficient orientation. Then come the solar collector schemes. A number of the expandable types of solar collectors are now in their final phases of development. These solar power units are principally of three types: (1) the sunflower petal-type having foldable petal-like panels which mechanically deploy and ck into place; (2) the umbrella type constructed of slender tubula. ribs attached to a continuous reflective membrane and deployed by the familiar opening movements of the umbrella; and (3) the inflated, foam-in-place, and foam-rigidized type of solar concentrator.

The inflatable paraglider using the Rogallo wing concept is shown in the next sketch on Figure 1. In general, this re-entry configuration consists of a high-strength flexible liftir surface with a leading edge of rigid or inflatable tube construction. All concepts shown beyond that of the paraglider are vehicles for manned occupancy with the possible exception of the inflatable, booster recovery system shown at the upper-third of the Figure. This recovery system has been propo. ed for both cargo and manned capsule recovery.

The first series of manned expandable structures shown in Figure 1 are trainer-type spacecraft using the Gemini capsule in conjunction with a semi-rigid telescoping module. The lower-most trainer depicts the first mission in which a simple-type expandable module is checked out. In this exercise, an astronaut may leave the confines of the re-entry capsule and enter the attached module to perform in-orbit evaluations of both the mechanics of the expansion process and the integrity of the related seals. The next class of trainer spacecraft depicted in the Figure shows a more complex expandable structure being employed in a rendezvous and docking exercise.

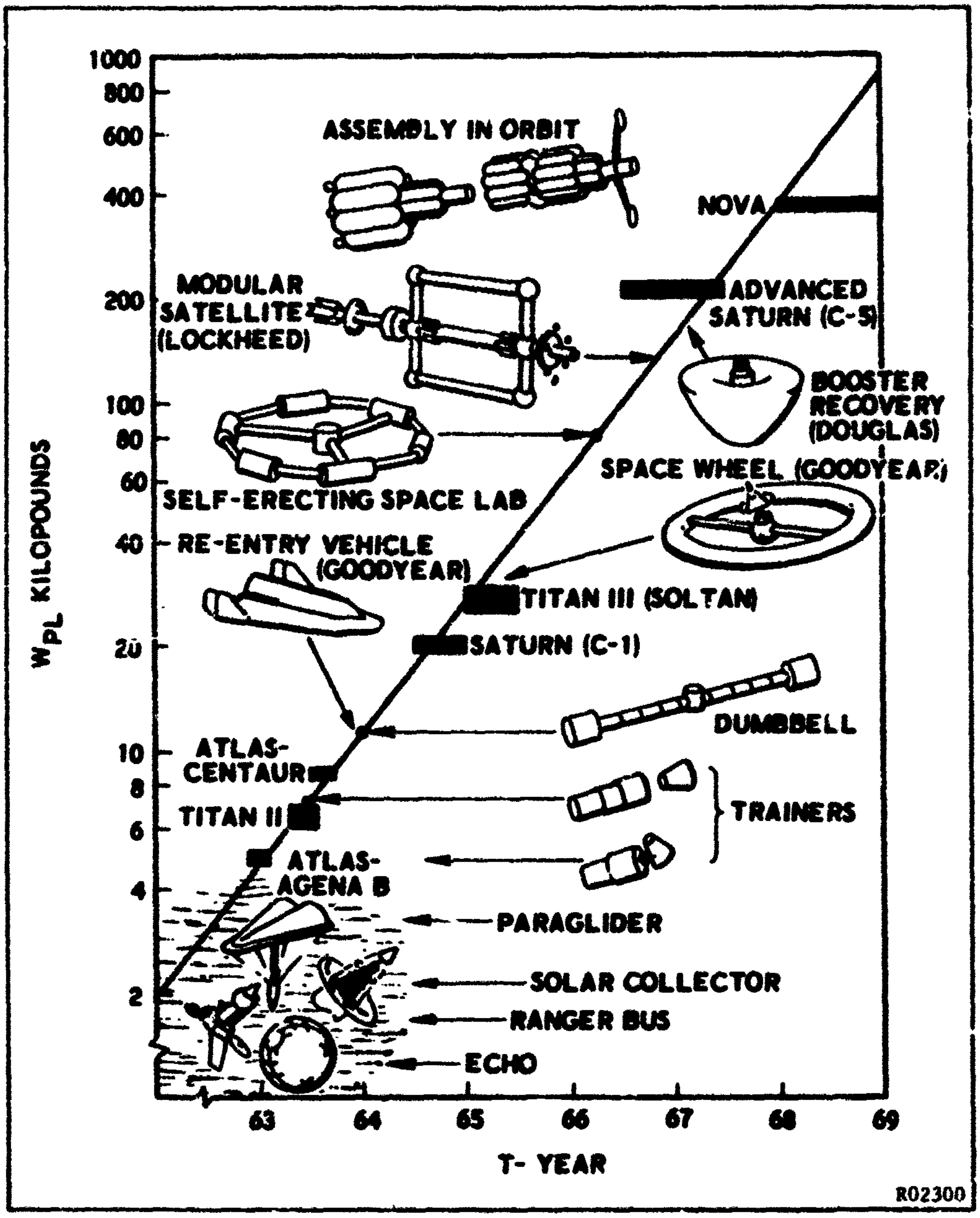

FIGURE 1. PAYLOAD CAPABILITY - 300-MILE ORBIT

The manned re-entry glider of the inflatable Dyna Soar type is shown in the next sketch. Inflatable vehicles of this type are under current study by Goodyear using the Airmat principal of construction. Airmat enables inflation into shapes other than that of bodies of revolution. The next three spacecraft fall into the self-erecting, manned space-station category. The two at left side of the Figure are of semi-rigid construction, and the Goodyear space station on the right side uses either flexible single-wall or Airmat construction.

Shown above the Goodyear space-wheel is the Douglas proposed booster recovery system which can be either manned or unmanned during recovery. Briefly, this system consists of a tremendously large drag cone for recovery of the expensive advanced Saturn or Nova boosters.

The final two manned spacecraft shown in Figure 1 are those of assembled-in-space concepts. For these very large and massive vehicles, prefabricated modules are proposed to be individually boosted into orbit and assembled. The lower of the two concepts shows Lockheed's modular space station which may also incorporate intermediate elements of expandable construction. The uppermost concept is typical of that now being proposed for initiating interplanetary missions. As indicated in this final sketch, the major assembly of such a vehicle is that of tankage clusters needed to store cryogenic fluids dictated by the propulsion system.

The several structural configurations displayed on Figure 1 are only a few of the many concepts that are now being proposed for missions in space. Nevertheless, the wide spread of concepts shown are adequate for the general discussions that will follow concerning typical designs and the related criteria for materials selection. It should be noted that the positioning of the concepts with respect to the time-scale indicates no more than payload capability and are not accurate estimates of the time for the first flight of these spacecraft.

PACKAGING EFFICIENCY

Packaging of an expandable structure is of major importance in the evaluation of a concept. Not only must such a structure meet the mission requirements, but it must also be capable of being collapsed and packaged, without damage, into a compact, efficient, boost envelope. Thus, there is a strong interaction between the structural complexity introduced by the mission requirements and the packaging technique. The packaging schemes for non-rigid, inflatable structures involve, in most instances, a trial and error process in which the principles of engineering have little application. It should be mentioned, however, at this point that considerable effort is being expended at Astro Research Corporation[1] in the study of theoretical folding patterns for the more idealized membrane structures. Semi-rigid structures, on the other hand, are packaged according to the nature of the sectioning of the structure which, in turn, is dependent on the overall configuration and its mission requirements.

The tight packaging of the Echo I and Beacon satellites enabled expansion ratios* of approximately 100,000 and 3,400, respectively, to be achieved. These extremely high ratios are representative of what can be expected with the thin-wall inflatable spheres; however, as both the load requirements and structural wall complexity increases the expansion ratios can be expected to diminish rapidly. The trend between expansion ratio and configuration complexity is emphasized in Figure 2. In this Figure, a number of the existent structures having available package dimensions are introduced. As indicated by the shaded region of the Figure, expansion ratios ranging from 2 to possibly 6 are about as high as one can reasonably expect for any type of manned expandable structure designed for extended periods of time in space.

NON-RIGID CONFIGURATIONS AND RELATED MATERIALS

DEFINITIONS

The non-rigid structures are considered to be those which are constructed principally of flexible materials. These structures may be folded or rolled into complex patterns for packaging without resorting to hinges or other specialized schemes for providing flexibility in localized regions. Non-rigid structures are necessarily non-rigid only in their collapsed form. Once erected they may either remain non-rigid with stabilization accomplished solely by inflation pressure or they may become rigidized after deployment by a number of techniques. Rigidizing schemes include: (1) mechanical systems - linkages, tie rods, etc.; (2) physical systems - elastic property changes, elastic recovery, etc.; and (3) chemical systems - foaming and rigidizing of foams, plasticizer loss, activation of resins, etc.

FLEXIBLE WALL CONCEPTS

Several flexible wall concepts under consideration for non-rigid vehicles are shown in Figure 3. The single-skin construction, displayed at the upper left, is shown with fabric reinforcement which enhances the strength, flexibility, and tear-resistant characteristics. These single-wall materials, with or without reinforcement, are generally placed in the category of balloon materials. They may range from the aluminumized plastic films as used on the Echo I Satelloon to the coated metal wire cloth proposed for re-entry drag devices and manned re-entry vehicles. The structural stability of configurations using the single-skin construction would generally be provided by internal pressurization; however, methods of introducing stiffness into membranes and coated fabrics are now under study. Westinghouse Electric Corp.,[2] is well along with research on glass fiber base fabrics and special flexible

* Expansion ratio = expanded volume/packaged volume

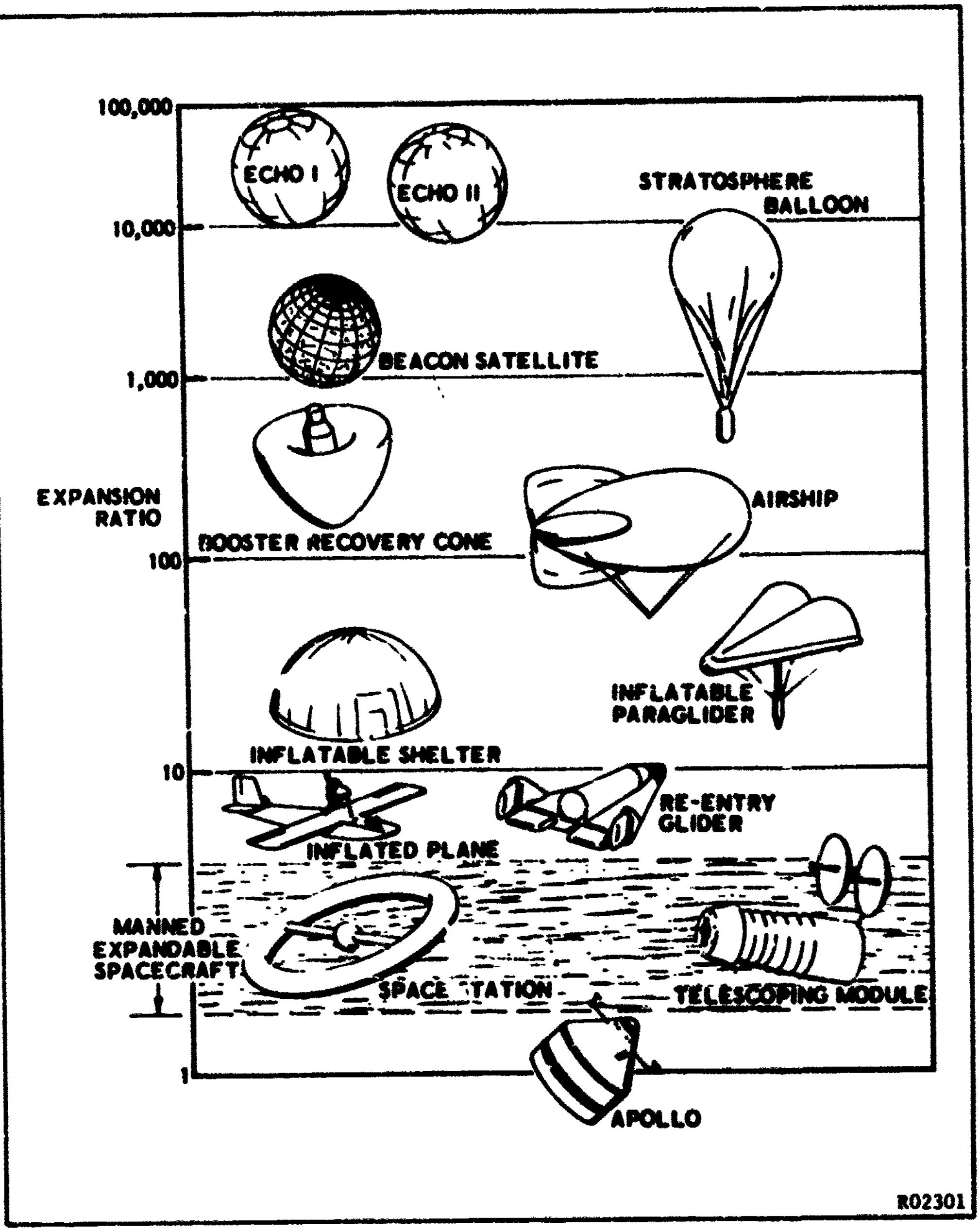

FIGURE 2. TRENDS IN EXPANSION RATIOS

coating formulas which lose their plasticizer after deployment and thus become extremely rigid. Hughes Aircraft Company and NASA are also developing methods for rigidizing balloon-like structures. Hughes has shown the feasibility of using two different methods for rigidizing thin wall laminates. One method relies on exposure to ultraviolet rays to rigidize a polyester resin combined with a dacron mesh laminated between two layers of aluminumized Mylar. The other method employs a thermal sensitive laminate composed of an epoxy resin coated over a fabric which has been faced with aluminumized Mylar. The NASA laminate under study consists of two layers of microthin aluminum foil with a 1/3 mil Mylar film sandwiched between. This system is then rigidized by overinflation so that it takes on a permanent set (tensile residual strain). All of these rigidizing schemes appear promising for huge balloon-like structures and giant solar collectors.

The Airmat* material, shown as concept (B) in Figure 3, is actually one integral piece of cloth woven into a double-wall configuration with continuous, closely-spaced drop threads regulating the dual-wall depth. The surfaces of the resulting fabric are then coated with suitable elastomeric materials to make the construction airtight amidst the design environment. The Airmat cloth has been available for several years in cotton, Nylon, Fortisan, and Dacron coated with butyl or neoprene elastomers; and now fiberglass and metal cloth are being successfully employed with heat-resistant coatings. The material is rigidized by intrawall pressurization, and the drop thread arrangement permits application to various shapes that would otherwise be impossible for the single-wall materials.

A foamed core material sandwiched between flexible facings is also shown in Figure 3. This wall concept, labeled (C), represents two distinctly different applications of foam materials. The first application is that in which a prefoamed flexible core material rigidizes flexible facings using the principles of expansion through elastic recovery mechanisms.[3] The second application of foam core materials is that of foaming-in-place between flexible facings after deployment, thus achieving the desired characteristics of a rigid sandwich.

In the former application, the flexible foam is prefabricated into the wall cross section prior to packaging. By compressing the foam core and firmly folding, the structure can be packaged into a relatively small volume container. Upon release from the container in space, the stored potential energy of the compressed core is sufficient to expand and lend rigidity to the configuration. No auxiliary forces such as gas pressurization, chemical reaction, or mechanical devices are required in the deployment scheme. A similar application is that of replacing the foam core with a flexible core fabricated of corrugated elastomeric materials such as neoprene, polyvinyl, reinforced films, etc.

* Goodyear Aircraft Corporation Trademark

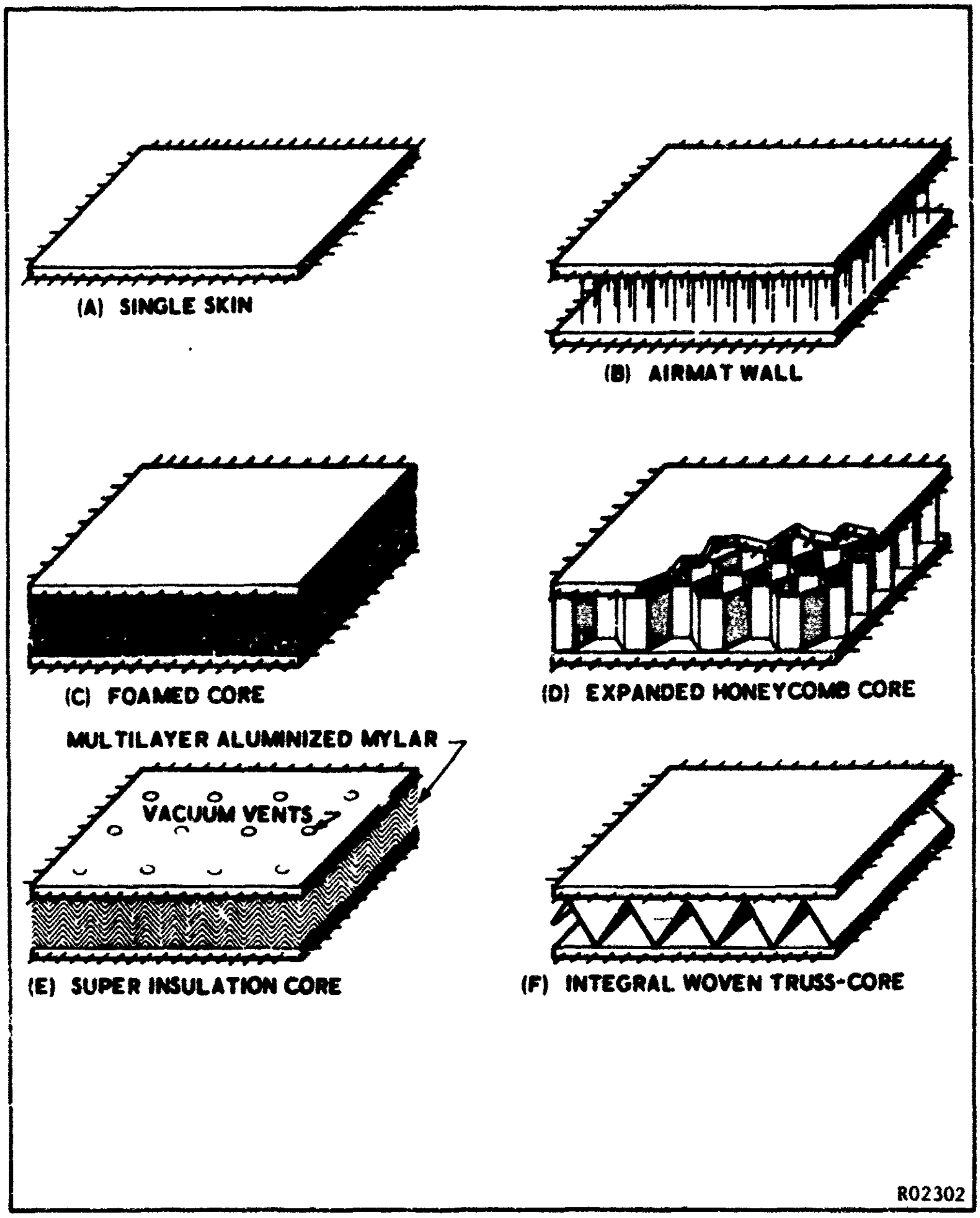

FIGURE 3. FLEXIBLE WALL CONCEPTS

The development of foam-in-place techniques for rigidizing structures is under way in a number of laboratories. The method receiving greatest attention is that which employs a foam reaction within the wall of a previously inflated double-wall structure. For this purpose, the polyurethane foams appear to be the more promising. Formulation and processing of these foam materials have taken the form of both powders and liquid reactants, and at the present time, a one part solid reactant system activated by heat alone is under development.[4]

The next wall cross section shown at (D) in Figure 3, is that of a double-wall construction having flexible coated-fabric facings and a honeycomb-like core. Schemes of this type are under study[5] whereby the core consists of a conventional metal honeycomb, and during deployment, the unexpanded core is expanded by inflation to a predetermined shape. The inner skin is used as a bladder to force the expansion while the outer skin serves as a restraining medium once the final shape is achieved. Methods of bonding the skins to the expanded core are being developed. These include: (a) rupturing by means of contact pressure of an encapsulated adhesive system attached to the skins, and (b) methods for local heating and melting of capsules containing an epoxy resin and a catalyst followed by curing into a suitable bond.

The double-wall concept, shown at position (E) in Figure 3, employs an intrawall insulation scheme which will find application in expandable manned spacecraft construction. This flexible wall concept employs vacuum-vented superinsulations which are contained between the facing skins. Since these insulating materials are non-load carrying, the outer vented skin can only serve to contain the insulation and to protect it from degradation. This scheme will either be used in conjunction with one of the other wall concepts or it will be applied to areas where the inner skin can handle the structural loads. The superinsulations typified by the Linde SI-4 and NRC-2 materials have extremely high insulative characteristics when vented to vacuum conditions, and the principal problem will be one of containment without the introduction of heat leaks through the insulation. The purpose of this type of construction would be to attenuate the large thermal fluctuations resulting from solar heating of the exterior surface, thus enabling a close control of the shirt-sleeve environment within a manned craft. Also, the concept has important application to the shrouding of cryogenic vessels for long-time storage in space.

The final sketch, (F), in Figure 3 displays an orthotropic core concept. That is, the core material is directional in core construction and strength characteristics. Several fabrication schemes for imparting flexibility into this type of wall construction have been developed; however, methods or rigidizing in space are still under study. An integrally woven, three dimensional fabric material (Raypan*) is now available in several wall depths, resin types, yarns, etc. Either truss core or vertical webs of the type shown are being manufactured. All of the methods mentioned earlier for rigidizing fabrics would be equally applicable for this construction. In

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* Registered Trademark of Raymond Development Industries

addition to the usual structural wall applications, the directional core construction would enable the wall to serve as a heat exchanger in critical areas of the vehicle. In this way, internal heat generated by power supplies, etc., could be carried to the outer-skin and radiated to space.

The specific applications of some of the wall concepts briefly described above are discussed in the following Section.

THE BALLOON SATELLITE AND THIN-WALL CONSTRUCTION

The Echo I satellite was successfully launched and deployed a little over three years ago. This achievement represented the first introduction of an inflatable structure into orbit, and it proved, without a doubt, the value of this inflatable concept for world-wide communications. The overall Project Echo[6] was managed by NASA's Goddard Space Flight Center, and the development and testing of the 100 foot-diameter satellite, itself, was under the direction of NASA's Langley Research Center.

Echo I was an extremely thin-wall structure in that it was fabricated from one-half mil (0.0005-inch thick) Mylar* (polyethylene terephthalate) coated with a vacuum deposited molecular film of aluminum. The aluminum coating provided the necessary reflectivity to radio frequency transmissions. The weight of the balloon, including about 5 miles of seams, as fabricated by the G. T. Schjeldahl Company, was about 104 pounds or approximately 3.4 pounds per 1000 square feet of surface. Within the balloon, 30 pounds of sublimating powders were carefully sprinkled so as to develop sufficient vapor pressure for inflation once deployed amidst the vacuum of space. The payload weight also included two miniature solar powered radio beacons (11 ounces) carried for tracking purposes. These items constituted the satellite vehicle which was carefully packaged within a magnesium canister of 26-1/2 inch diameter. A Thor-Delta vehicle was used to launch the package into orbit. Both the package arrangement and the method for initiating deployment of Echo I are shown in Figure 4.

Although this balloon-like inflatable structure presented a construction of simple form, it was far from being free of problems, especially when such balloons come in 100-foot-diameter sizes. The deflated membrane structure extended 157 feet across the work table, and it presented difficulties in fabrication, handling, inspection, testing, and packaging. Fortunately, however, experience gained in the past decade with stratosphere balloons, and more recently, with superpressure balloon systems enabled many problems to be anticipated and eliminated before they hampered the program. For example, experience was gained with thermoplastic strip adhesives using polyester resins during the development of superpressure balloons, and this experience was brought to bear with little difficulty in the fabrication of all seams on Echo I.

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* Registered Trademark of E. I. duPont de Nemours and Company

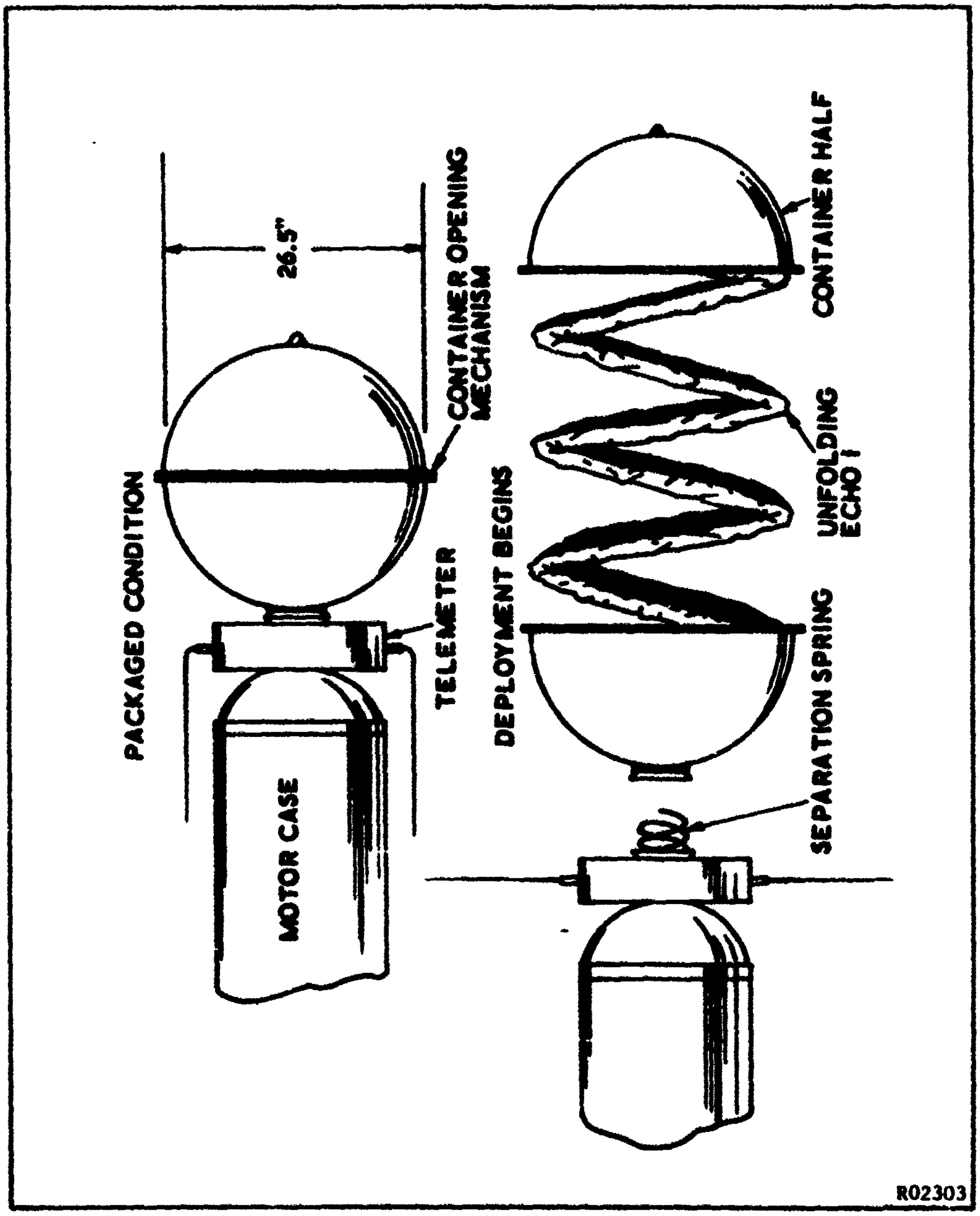

FIGURE 4. ECHO I PACKAGE AND DEPLOYMENT

New problems in packaging, beyond that experienced with atmospheric balloons, did present themselves, however. In the packaging of Echo I prior to launch, it was necessary to remove all residual air in the balloon by piercing it with about 250 strategically placed pin holes and then placing the package in a vacuum chamber to draw out the entrapped air. While in this evacuated condition, the package was vacuum sealed into a spherical canister. Were this not done, local air pockets would undoubtedly have produced uncontrolled inflation and localized ruptures would have scuttled the satellite. Controlled inflation was relegated to the subliming powders. These were carefully selected to give exactly the range of vapor pressures required as the satellite passed in and out of the earth's shadow. The magnitude of the equilibrium pressure developed within the satellite was directly dependent, for the most part, on the magnitude of the membrane skin temperature. Therefore, since the skin was subject to large temperature variations as it passed through the darkness (-157°F) and sunlight (+250°F) portions of the orbit, it was extremely difficult to keep the internal pressure below the bursting pressure (approximately 0.03 psi) in the sunlight region while still retaining sufficient pressure in darkness to prevent compressive instability from the external pressure exerted by the combination of earth radiation and stagnation pressure.

For the Echo I membrane design, it was concluded, however, that the rigidity of the satellite was sufficient to retain its shape during earth orbit at 1000 miles altitude.[7] On the other hand, at somewhat lower orbit altitudes where external pressures become more significant, a slight measure of increased rigidity would be required to prevent buckling instability and the accompanying drop in reflective efficiency. This need for more rigidity in thin-membrane inflatables has prompted several research programs to develop rigidizing schemes. These were briefly discussed in the preceding Section. The NASA laminate which consists of the two layers of aluminum foil (0.00018 inch) with an interlayer of thin Mylar film (0.00035 inch) was used in the larger Echo II Satellite (135 foot diameter). It is estimated that Echo II (also known as Echo A-12) will weigh twice that of Echo I and will require a proportionately larger package.

Another technique for rigidizing reflective spheres like Echo is that of a wire-rigidized, aluminized, Mylar material under study by Goodyear. This scheme also uses the "over-stress" principal to develop a residual tensile strain in the wire mesh. The rigidity thus developed by overinflation is then high enough to retain the shape in low orbits even after all pressure has been lost. An auxiliary technique that has been proposed is that of subsequently evaporating away the plastic film covering the mesh. This method would leave only the open fine-wire frame, thus reducing the adverse effects of radiation and stagnation pressures. The wires are still close enough, however, to cause little reduction to its efficiency as a radio wave reflector.

THE FLEXIBLE SOLAR COLLECTOR AND FOAMED CORE CONSTRUCTION

One of the more attractive methods now being developed for erecting large solar collectors in space is that of inflation followed by rigidization using foam-in-place techniques. Goodyear Aircraft Corporation has developed a rather straightforward procedure for accomplishing this type of reflector. The precise shape of the reflector is attained by pressurizing a volume located between the concave surface of the reflector and a temporary hemispherical forward section. The aluminumized reflector surface is then rigidized by releasing a lightweight foam in a double-wall cavity immediately behind the reflective surface. Finally, after sufficient time has elapsed to allow the foam to cure (typically, 15 to 20 minutes), the hemispherical forward section is removed along the rim of the reflector leaving the desired dish-type configuration. The sequence of events just described are clearly shown in Figure 5. Huge solar collectors up to 50 feet in diameter are in the planning stage; however, at the present time smaller "feasibility" models are under construction to improve foaming techniques and fabrication methods.

The polyurethane family of foam materials are receiving most attention for such foam-in-place applications, especially in the presence of the space environments. Their popularity stems not only from their foaming characteristics but also from their resistance to degradation (after foaming) by the combined environment of thermal radiation and vacuum. The polyurethanes retain good strength properties up to temperatures near 300°F; however, some protection from ultraviolet rays is required for long-time exposures. This shortcoming is not severe since protective coatings have been developed and successfully used against ultraviolet rays.

The Mylar films, as in the Echo balloon application, appear to satisfy the requirements for the membrane surfaces of the inflated and foam rigidized solar collector. Film thicknesses of 1/2 to 1 mil are in general use in most developments of this type. A vacuum deposit of aluminum is, of course, added to the Mylar at the reflector surface. For normal use in space, this aluminum film needs to be no more than about 2000 angstroms thick (8.0 micro-inches).

The conventional polyurethane foams, referred to above, foam satisfactorily under the moderate vacuum conditions existing within the confines of double-wall membranes. From the standpoint of vacuum, these intrawall conditions are not severe, although these regions have received the standard evacuation procedures before packaging. Unfortunately, however, in the hard vacuum (10^{-8} to 10^{-11} mm Hg.) existing outside a deployed space structure, the conventional foams cannot be used since they will collapse during the foaming process. Therefore, methods for foaming a material that adheres to the vacuum-exposed outer surface of a structure requires new material formulations and processes. A number of research programs for foaming under the difficulties of hard vacuum are underway, and promising results are beginning to be displayed. Hughes Aircraft Company has developed a powder mixture of solid urethane reactants which is made to adhere to the exterior of an inflatable structure. Once it reaches a temperature between 180° and 250°F a foam reaction is triggered into operation.

FIGURE 5. FLEXIBLE SOLAR COLLECTOR

Gelatin films are being studied at ASD as a possible means of rigidizing structures of the solar-collector type. The gelatin film is cast onto flexible surface and allowed to cure to the point of being flexible but not sticky. Then, the structure is folded and packaged into an evacuated container which prevents rigidization of the film by preventing evaporation of water from the gelatin. Once deployed in space, however, the water evaporates and the film rigidizes. Furthermore, as soon as the rigid surface reaches a temperature of about 200°F, the film foams to a thickness ten or so times its original thickness, thus imparting additional rigidity to the surface.

Another method, that of spraying foam reactants automatically against a surface exposed to the space environment, has been under development by Goodyear. In this scheme a "back flap" method is employed whereby the foam is applied between a flap and the exterior surface. Thus, the foaming actually takes place, and is allowed to quick-cure, in the protective confines underneath the flap. Beneath the flap, the exterior hard vacuum has not had time to penetrate amidst the entrapped gases. It is reported that this system may be used to rigidize a prototype solar collector soon to be placed in orbit.

THE FLEXIBLE PARAWING RECOVERY DEVICE

The flexible recovery devices referred to as either parawings, paragliders, or Rogallo wings represent a considerable advancement in maneuverability over the steerable parachute. However, the greater maneuver capability is provided at the expense of more complexity in the structural arrangement and an increase in the aerodynamic problems. Nevertheless, the merits of the concept overshadow its difficulties to sufficient extent that considerable development in this area can be expected in the years ahead.

The principal structural members employed in these recovery configurations are a central keel and two sweptback leading edges. Typically, these are tubular members that are rigidized by inflation at the beginning of the deployment sequence. A large-area, flexible, lifting surface is attached continuously along the length of the tubes. This aerodynamic surface, generally, consists of a very flexible, high-strength fabric coated with a thermal resistance coasting. The deployed paraglider shown in Figure 6 is representative of the construction just described. Although the vehicle shown is that of a micrometeoroid sensing paraglider, it is very similar to most of the other proposed systems including that being developed for Gemini recovery.

The micrometeoroid paraglider was developed by the Space General Corporation and B. F. Goodrich Company under the direction of NASA/Langley. The paraglider is to be boosted to an altitude of 250,000 feet, where it is deployed. It then will follow a nominal trajectory and glide path during which time micrometeoroid impacts on sensors will be detected and telemetered back to earth. The materials and processing problems encountered in the development of the vehicle are typical of this type of aerospace configuration; however, due to the nature of the experiment to be performed, some added problems related to materials may be considered unique. That is, the scheme for measuring the micrometeoroid flux in near space required that all of the available 110 square feet of lifting surface be used for mounting

sensitive capacitor-type sensors. These sensors were mounted on both sides of the surface giving approximately 200 square feet of sensor area. The very low aerodynamic loading on the lifting surface enabled a very lightweight, single-ply, fiberglass cloth to be used for this application. The material used was 3M-Y9050 with an adhesive applied to both sides to simplify the mounting of the sensors. This material was available in 36-inch widths, and 2-inch lap joints were employed in building up the required surface area.

The capacitor sensors consisted of six layers of aluminum film, each 25 micro-inches thick, interleaved with three layers of 1/4 mil Mylar and two layers of 1/2 mil Mylar to form a multilayer capacitor about 0.002 inch thick. This laminate was bonded together by heat and pressure using a Schjeldahl 301 adhesive system. To assure adhesion under all temperature extremes, the laminate had to survive a thermal shock test involving rapid transfer from boiling water to liquid nitrogen.

The basic material of construction selected for the tubular structure was Hess Goldsmith HG66 fiberglass fabric. This is a 17 by 17 plain weave material having a thickness of about 0.016 inch. Its nominal breaking strength is 600 lb/inch. Two layers of this precoated fabric were orthogonally matched and orientated on a bias with respect to the longitudinal axis of the tubes. The matrix material was selected to be a silicone elastomer because of its superior high-temperature characteristics and flexibility. Low permeability was not a primary requirement since a low permeability film could be applied on the inside of the tube if excessive loss of gas were encountered during proof testing. The basic strength required of the fabric material was dictated primarily by the level of internal pressure needed to prevent buckling of the tubular members under the design loads.

During the development of this vehicle, some weakening and breakage of the glass fibers in the fabric was noted as a result of the coating and handling operations. This characteristic weakening appears to be common to all composites using fiberglass as a flexible reinforcement, since it is made from filaments that are basically brittle. The situation is further aggravated by removal of sizing to improve the adhesion of silicone. Sizing acts as a lubricant between adjacent fibers, and when this is removed, friction and breakage begin. These are problems that have been recognized, and they should be carefully considered in design. It appears improbable that fiberglass materials can be used at their full rated strength if they are to be flexible.

As indicated earlier the lifting surface area of the paraglider in Figure 6 is 110 square feet. The structural tubes are tapered, as shown, from 20 inches to 8 inches over their 12-foot length. The vertical inflated tube is 12 inches in diameter and extends to 12 feet with a rigid hardware canister attached at the lowermost end. The canister and hardware weigh 90 pounds, and the flexible paraglider weighs 90 pounds for a total package weight of 180 pounds. A fixed trim is employed for the vehicle, and a peak wing loading of approximately 2.5 lb/ft^2 has been predicted. The packing density (package volume/calculated material volume) has been established at a value of 3.0 for the flexible portion of the package.

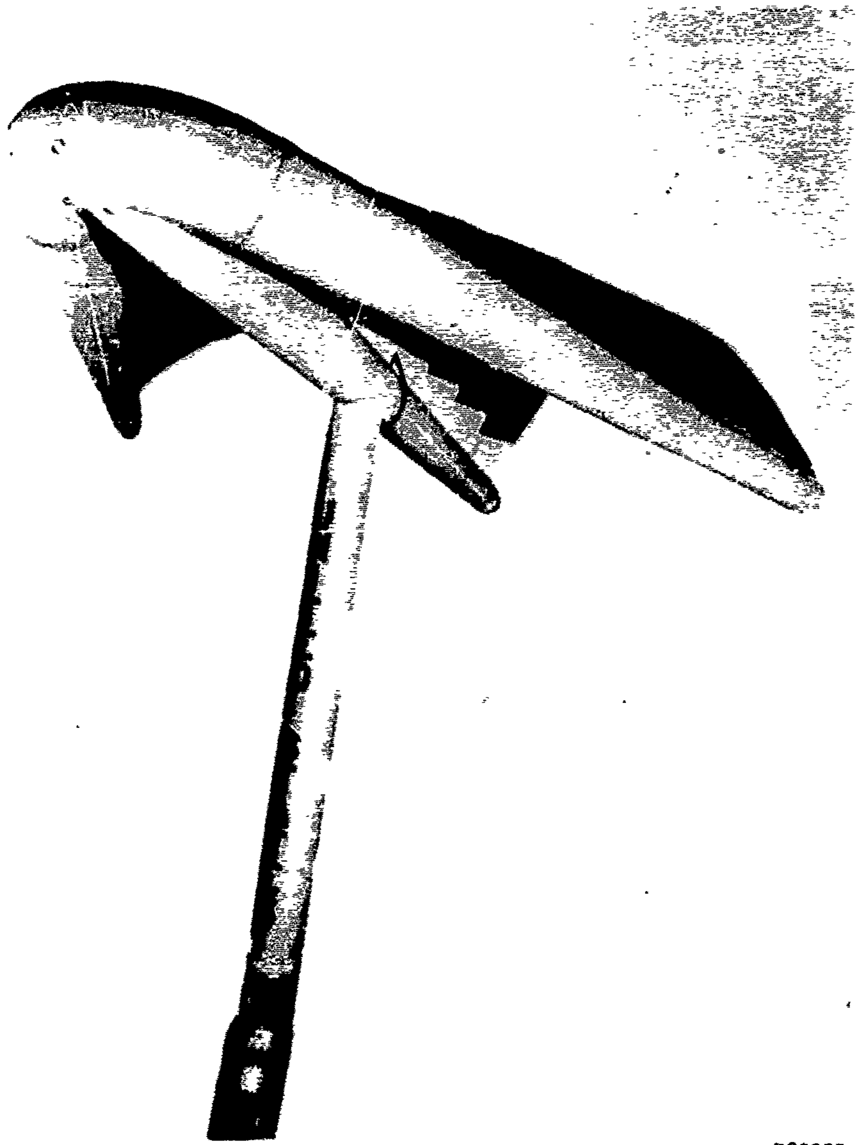

FIGURE 6. THE MICROMETEOROID PARAGLIDER

THE INFLATABLE RE-ENTRY GLIDER

The so-called thermal barrier associated with aerodynamic flight creates the basis for a pressing structural problem during glide re-entry. Gliders of the Dyna-Soar type, for example, with their relatively high wing loadings and high re-entry velocities will have structural temperatures ranging between, say 4000°F at the stagnation point on the nose to about 2500°F underneath the wings. Typical relationships between velocity, altitude, wing loading, and induced temperature for glide re-entry are shown in Figure 7. The lowermost wing-loading curve is that of a conventional glider having a wing loading of 80 lb/ft^2. The temperature lines shown are representative of those occurring beneath the wing. A maximum temperature between 2000°F and 3000°F is indicated for the conventional vehicle. Furthermore, the severity of these temperatures for the re-entry glider are emphasized by long-time exposures which are of the order of minutes. Thus, for the conventional glider rather exotic materials are indicated to cope with the thermal problem.

A reduction of the wing loading on the glider has the effect of reducing the level of aerodynamic heating. This is indicated in Figure 7 by the uppermost wing-loading curve typical of gliders having large wings and thus lower velocities for a given altitude. The conventional glider, however, is restricted to wings of low aspect ratio in order for the configuration to be compatible during the exit phase with the stability and control capability of the booster system. In turn, these dimensional constraints restrict the glider to the higher wing loadings and temperatures. It is here that the inflatable glider appears to offer advantages. By using the inflatable concept, wings of large size can be folded around the fuselage during exit and then inflated before re-entry, thereby achieving the desired wing-loads of low magnitude.

The inflatable concept publicized by Goodyear, is shown in Figure 8. The Airmat material described earlier and shown in Figure 3 is used extensively in this vehicle. The wing loadings, typical of the proposed inflatable gliders, range between 1.0 and 5.0 lb/ft^2; and, as shown in Figure 7, this corresponds to re-entry temperatures of about 1700 to 1500°F. In this temperature range, a number of less exotic materials have good structural characteristics. In fact, fabrics woven of materials such as the austenitic stainless steels, nickel-base alloy steels, and fiberglass show promise.

Unfortunately, the application of expandable concepts to a glide re-entry vehicle, as with any other vehicle, is more than just a basic materials problem. The broad problem is not only one of developing materials with good, elevated temperature strength, low permeability, etc., but it is also one of developing new methods of fabrication and refined methods of predicting the structural response of the new fabric-base materials. For the glide vehicle, in particular, the flexible structure must be capable of not only holding a prescribed geometrical shape, such as that of a fuselage or an air-foil, but it must also provide a good, predictable measure of protection to the crew while subjected to the hazards of space and re-entry.

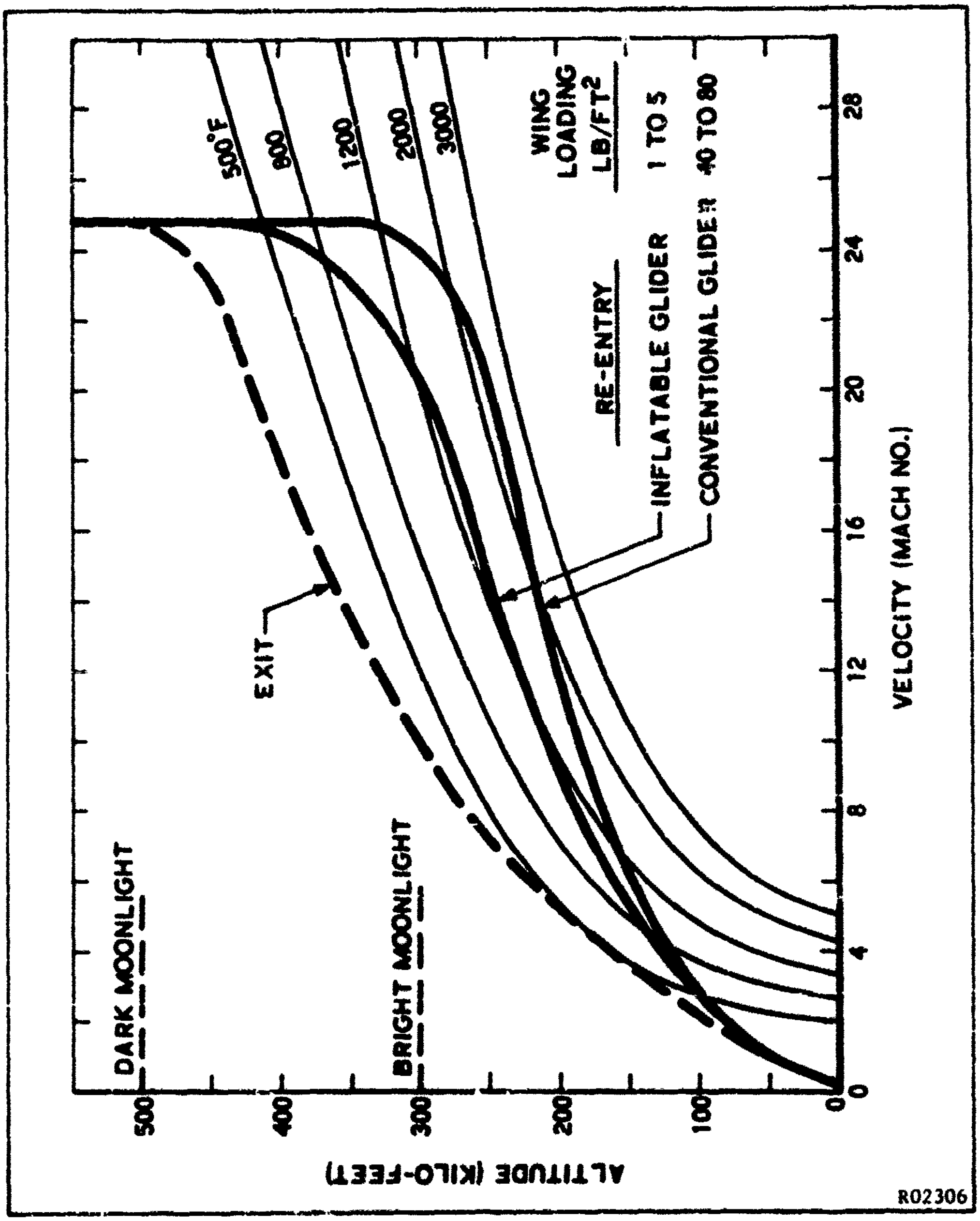

FIGURE 7. AERODYNAMIC HEATING VERSUS RE-ENTRY CORRIDOR

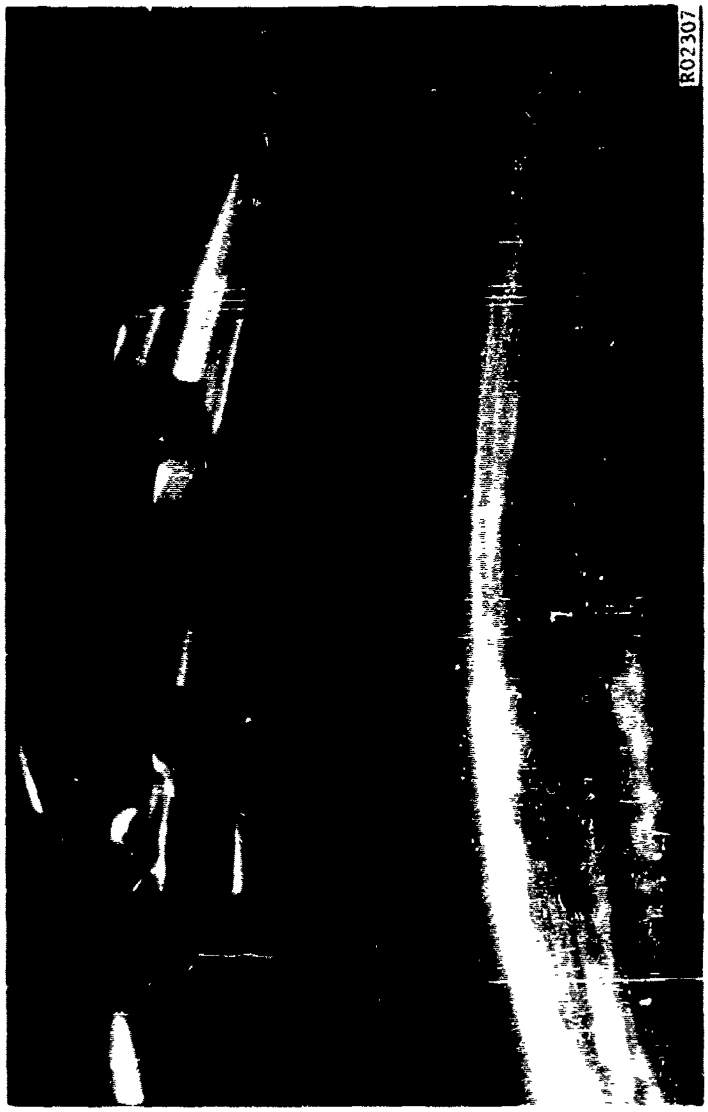

FIGURE 8. INFLATABLE RE-ENTRY VEHICLE

Undoubtedly, the inflatable space station concept that has received the most extensive study is that of the inflatable torus fabricated by Goodyear. As in most space station concepts, the tore-type of spacecraft is designed to develop an artificial gravity through rotation about its own axis. Due to human factor criteria,[9] this means of inducing gravity also restricts the minimum diameter of the rotating station to about 80 feet.* Although a full scale vehicle has not been fabricated, Goodyear has built several experimental structures (1/2 size and smaller) for use in a number of studies. Problem areas under study include fabrication methods, variable gravity effects, folding, erection, internal environment, system dynamics, etc. A 30-foot diameter experimental torus structure is shown in Figure 9. The test vehicle shown here is supported at the hub to allow inspection and rotation during the study. (Note the use of tie cables and shear-pad attachments for increasing the rigidity of the structure.)

The vehicle shown in Figure 9 is fabricated of Airmat material which gives the desired feature of double-wall construction. The use of Airmat enables the inflation pressure of the structure to be independent of the pressure requirements placed on the internal environmental air. Futhermore, the dual-wall configuration affords added protection from meteoroids by presenting an effective "Wipple meteor-bumper" shield. This type of construction is also amenable to intrawall foaming procedures so as to rigidize the station after the initial inflation. It is feasible that the foam-core material could serve as an efficient attenuator of penetrating radiation as well as a means of improving structural integrity by eliminating the dependence on intrawall pressure for stability. Several fabric and coating schemes for the Airmat are under evaluation. These include fabrics of Nylon, Dacron, and Fortisan, and coatings of neoprene, butyl, and Mylar films. Improved methods of structural analysis of Airmat-type materials have also been developed in parallel with the fabrication programs. Much of the initial theoretical work was performed under the direction of NASA-Langley;[10,11] however, recently the Air Force has concentrated efforts[12] in these areas. In all of these studies the theoretical developments have been compared with experimental results.

Single-wall construction consisting of either multiple coated fabrics alone or filament windings combined with coated fabric liners has also been given some consideration for space station applications. Although the strength and permeability requirements can be met by these single-wall materials, it is doubtful that sufficient protection from environmental hazards can be given crew members for the extended lifetime of the space station and the typical periods envisioned between crew rotation.

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* Human factor considerations indicate that the differential gravity force from head to foot of an astronaut standing within the rim of the torus should be no greater than 15 percent; thus, for a 6-foot man, a minimum of 40-foot radius is indicated.

The merits of the inflatable glide re-entry vehicle has prompted extensive research and development in all aspects of fabric-base materials applications. A number of organizations are studying one or more of the problem areas concerned with high-temperature structural fabrics, coatings, manufacturing processes, and the applied mechanics problem. A partial listing of these organizations is as follows: Dow Corning, DuPont, Goodrich, Goodyear, H. I. Thompson, A. D. Little, and Westinghouse Electric. The majority of these people are working under Air Force contracts or subcontracts; however, in several instances the work is of a proprietary, inhouse nature. For reference purposes, several specific contracts and prime contractors are given below:

(1) AF 33(616)-8259, "Development of Fabric Base Materials for Space Applications," Westinghouse Electric Corporation.

(2) AF 33(657)-8702, "Analytical and Experimental Investigation of Coated-Metal Fabric Expandable Structures for Aerospace Applications," Goodyear Aerospace Corporation.

(3) AF 33(616)-7854, "New and Improved Materials for Expandable Structures," Goodyear Aerospace Corporation.

(4) AF 33(616)-7294, "Investigation of Properties of Fine Filaments and Fabrics of Superalloys and Refractory Metals," Arthur D. Little, Inc.

The information being generated by these programs should be recognized as being very preliminary in nature, and it would not be appropriate to compare, or even list, results obtained on material properties or methods of analysis at this early stage of their research. However, progress reports are available in which great quantities of information are unfolding. These should be of value for the preliminary trade-off studies.

THE INFLATABLE SPACE STATION

Space stations with large internal volumes and big working spaces for crew members are in the planning stage for future spacecraft. As yet, there is no approved national program for such an undertaking; however, the space-station concept is presently undergoing an intense study[8] to determine the most effective time to introduce this logical next step.

Goodyear Aerospace Corporation has indicated that a manned space station could be placed in orbit in about three years. The feasibility of the space station has been demonstrated by studies being performed by several NASA centers and the Air Force, as well as by a number of contractors in the aerospace industry. In general, it has been emphasized that the required space-station technology is less demanding than that of Project Apollo. In particular, complex areas such as navigation, guidance, reentry heating, propulsion subsystems, and lunar landing would be either absent or less stringent in their demands on the space station.

FIGURE 9. EXPERIMENTAL MODEL OF AN INFLATABLE SPACE STATION

EXPANDABLE BOOSTER-RECOVERY SYSTEMS

The reusability of advanced booster systems typified by Advanced Saturn and Nova may become decisively important to the logistic support given space stations and lunar bases in the not too distant future. Such recovery schemes appear impractical, however, with our present multistage booster systems due to the relatively low cost of the Atlas and Titan first stage boosters and the high cost of developing a reliable recovery system. Nevertheless, the picture changes drastically, once the booster becomes an expensive, one stage vehicle having 12 or so engines of large size; and, the initial costs of the recovery system are amortized over the entire life of the reusable boosters. These considerations have prompted a number of aerospace companies to make a careful evaluation of the cost trade-offs involved.

Thus, many expandable recovery schemes have come under study. These include parachutes, integral inflated wings, paragliders, and inflated drag cones. The latter method, namely, the inflatable, blunt-body drag cone shown in Figure 10 has been under study by Douglas[13] for several years. This is an extremely large inflatable structure whose diameter of 325 feet and height of 226 feet makes it several times larger than the inflatable space station referred to earlier. The recovery system, as proposed by Douglas, weighs approximately 107,000 pounds and would be composed of two distinct structural sections.

A "wrap-around" nose section is to be composed of an Airmat material using either Rene' 41 or 304 stainless-steel wire cloth impregnated with a high temperature silicone elastomer. The Airmat-type material is used to assure that the required conical nose shape will be retained. It is proposed that this nose portion be expendable; thus, the materials in the Airmat would be subjected to the 1500°F re-entry conditions only once. Prior to deployment, the Airmat nose section would be packaged in an aft canister centrally positioned amongst the engine thrust chambers.

The second section of the drag cone is proposed to be a single-wall coated fabric made from Dacron and coated with a silicone of low permeability. This would be a reusable section that is stored in an annular compartment at the aft edge of the booster skirt extension. When deployed, the single-wall section would be protected by the Airmat heatshield. It would take the shape of a large torus (8 million cubic feet) having a central, cylindrical pocket for holding the booster securely in place. Douglas has given the appropriate name, ROOST, to the systems concept.

The drag cone has a built-in "sky hook" buoyancy feature that enables the recovered booster and cone to become aerostatically buoyant at 2000 feet altitude. This feature is developed in the system by flowing a hot gas (hydrogen or helium at 400°F) into the inflated drag cone by means of a combustor-heater handling 25 to 55 pounds of gas per second. Gradual descent of the vehicle from the 2000-foot buoyant altitude to earth occurs in about 15 minutes as the bag cools off. Impact is no more than 2.5 fps. Without the buoyancy feature of the hot gases, impact would be 130 fps or higher. Schemes for making recovery on either land or sea, including the operations for returning the vehicle to the assembly area, have been studied rather completely by Douglas.

FIGURE 10. BOOSTER-RECOVERY DRAG CONE (CCURTESY DOUGLAS AIRCRAFT)

Minneapolis Honeywell is presently under contract to NASA to study a somewhat different concept for the hot-air balloon recovery system. This system, as presently conceived, resembles a large open-mouth super-pressure balloon 300 feet high and containing about 15 million cubic feet of hot air that rises from heaters into the open balloon. The recovery system is to consist of a parachute deceleration scheme, an array of gas burners for heating air to inflate and sustain the balloon, the large balloon itself, and either an automatic or remote control system to regulate both the sequence of events and the gas burners that heat the air. The final recovery procedure is similar to that of the Douglas concept in that descent is controlled by regulation of the inflation air temperature. The system can be towed to a suitable recovery area using helicopters, according to Honeywell.

SEMI-RIGID CONFIGURATIONS AND RELATED MATERIALS

DEFINITIONS

Semi-rigid expandable structures may be classified as those configurations which change either shape or volume by movements of rigid panels, sections, or modules. This classification includes both manned and unmanned concepts ranging from those of pivoting panels in the sunflower solar collectors to those of self-erecting space stations composed of both rigid modules and flexible interconnecting structure. Due to the predominance of rigid elements, the semi-rigid structures are packaged according to the nature of their sectioning, and, in general, the expansion ratios (as defined earlier) are small, that is, less than 10.

The rigid members of the semi-rigid concept may be of either metallic or non-metallic construction. The final choice of structural materials will usually be based on a minimum weight criterion; however, the familiar strength-to-weight relations may be overshadowed by considerations of environmental effects or structural instability. Furthermore, for the manned vehicles, major problems to be considered will be that of providing wall cross sections that show a good measure of resistance to meteoroid damage, shielding against penetrating radiation, and adaptability to passive and active thermal control.

RIGID-WALL CONCEPTS

The rigid-wall concepts that may be ascribed to expandable spacecraft will, of course, vary considerably in complexity depending on their structural and/or protective function. Probably the least complex wall section would be that of the deployable petals on a sunflower solar collector. These petals may be constructed of nothing more than a sheet of single-thickness material having a highly reflective surface coating. A somewhat more complex structural requirement is that of the hinged panels for supporting arrays of solar cells in the scientific satellites typified by Mariner, Ranger, Surveyor, Voyager, and Nimbus. In these latter applications, emphasis is placed on finding structural panels which display high rigidity-to-weight characteristics. As a result, most of the solar-cell panels now in use are constructed of honeycomb or truss-core sandwiches.

The extremes in rigid-wall complexity will undoubtedly be associated with the space-station modules within which crew members will be asked to reside for long periods of time. In the manned module application, it appears imperative that some form of multiwall construction be provided for handling the dual role of structural load-path and environment attenuation. A number of such composites are shown in Figure 11. The first composite shown is a typical wall concept in which polymeric materials have been used in combination with metals (or possibly rigid, reinforced plastics) to give an efficient design. Either a prefoamed polyurethane (1.0 to 1.5 cu ft/lb) or a super-insulation is employed here to serve several functions, namely, thermal insulation, meteoroid bumper, and radiation shield. Some structural rigidity would also be introduced by the foam even though the structural function will be primarily relegated to the sandwich construction. The merits of lightweight foam materials used in various meteoroid bumper arrangements have been studied extensively by General Dynamics, Goodyear[14] and Normco[15], and these materials hold good promise for meteoroid protection schemes.

The other rigid-wall concepts in Figure 11 are shown, for simplicity, without the outer protection system. For some applications, this may be all that is actually needed. That is, the primary structure may be able to give the desired degree of protection from the hazardous environment with only a slight off-optimum structural (load carrying) arrangement. By constraining the cross-section dimensions to meet the requirements for penetration resistance (meteoroids[16] and radiation[17]), and by using an orthotropic core sandwich as a heat exchanger, it may be possible to handle both structural and protective functions efficiently. Parametric design charts for such "constrained designs" have been developed by Aeronutronic for flat panels[18,19,20] and cylindrical shells[21] having single-truss, double-truss and corrugated-core sandwich construction.

Each of the wall cross sections shown in the Figure will display "efficiency factors" that vary considerably with the magnitude of the environmental constraints. Therefore, depending on the design loads and level of protection desired from the environment, these wall concepts may interchange their "efficiency" ranking.

SCIENTIFIC SPACECRAFT AND SOLAR PANELS

The unmanned scientific satellites typified by Mariner, Ranger, Surveyor, Voyager, and Nimbus have shown the importance of changing either size or shape in order to deploy arrays of solar cells, and thus, benefit from the "free-energy" supplied by solar radiation. In these craft, the need for adequate power for long-term operation of scientific gear, has dictated the use of deployable concepts. Structural deployment must be used regardless of the fact that such structural complexity inherently introduces problems in both reliability and structural response. A typical scheme for solar panel deployment, as used on the Ranger spacecraft, is shown by an artist's conception in Figure 12. In these semi-rigid scientific craft, the need for structural panels having high strength and high stiffness to support the solar cells and related equipment is apparent. Sandwich construction appears to satisfy the structural requirement; however, the introduction of joints and fittings into these sandwich materials, and

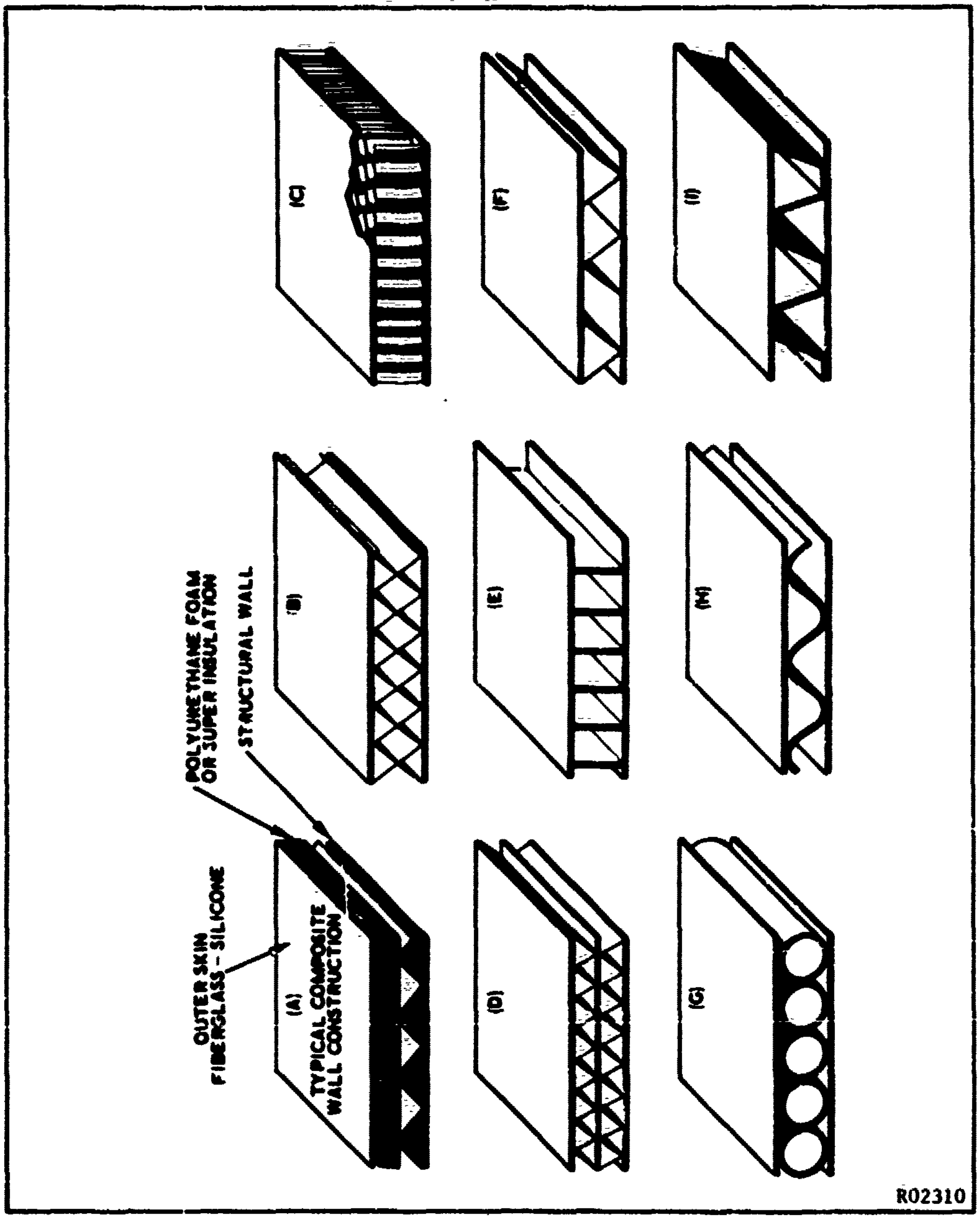

FIGURE 11. RIGID COMPOSITE WALL CONCEPTS

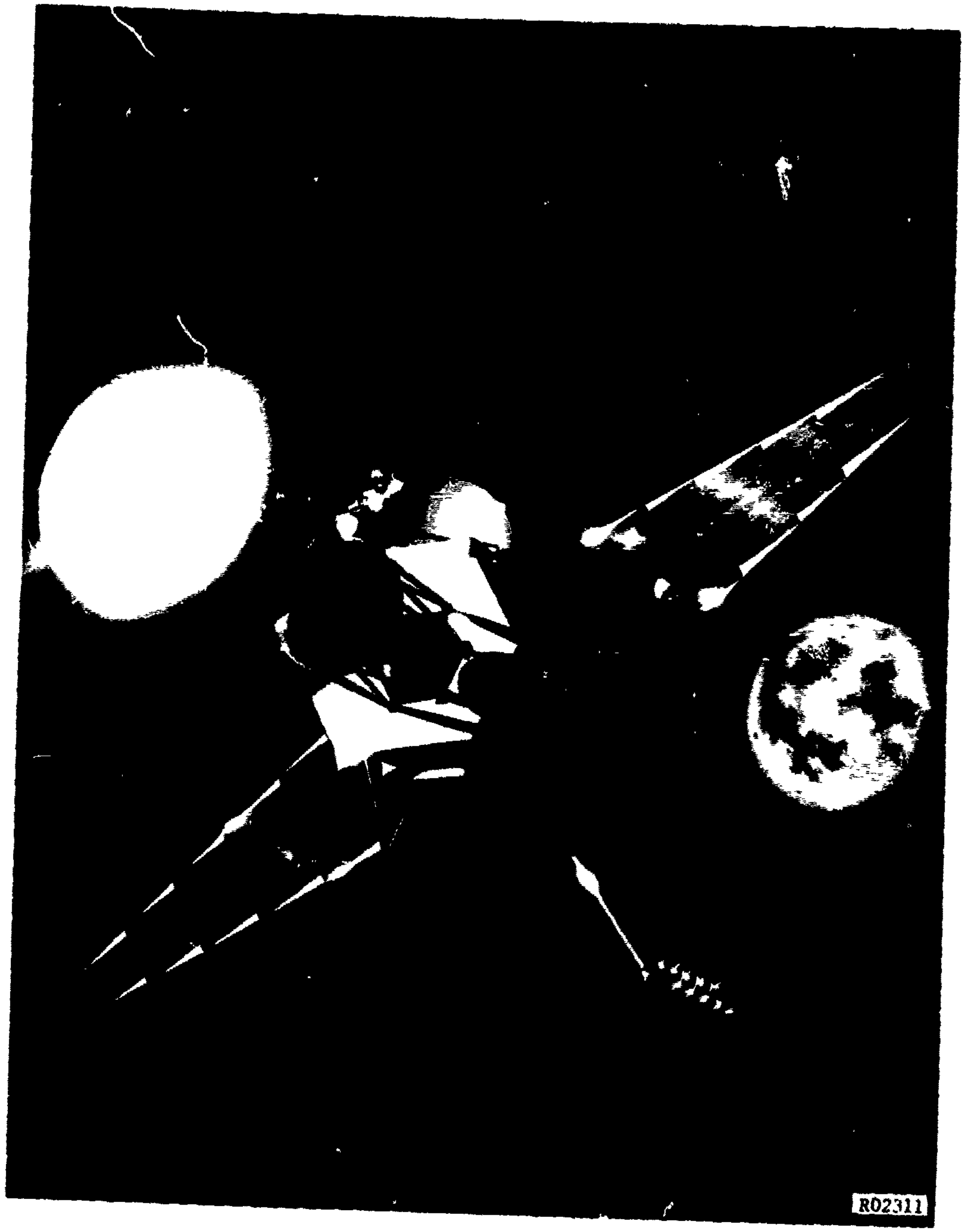

FIGURE 12. SOLAR PANEL DEPLOYMENT ON RANGER

the overall problem of bonding the facings to a core,hinders a straightforward development of panel designs. That is, the added weights associated with joints, fittings, and adhesives in sandwich materials can often become an appreciable percentage of the overall panel weight.

The truss-core and honeycomb-core sandwich designs displayed earlier in Figure 11 are now being applied to solar panel construction. However, the single-facing sandwiches (open-face type), having stiffening comprised of either integral waffles formed by the chem-mill process or built-up schemes, are attracting attention. The development of the solar panels for Mariner 2 is typical of the lightweight construction that can be achieved. These structural panels were built by Ryan Aeronautical, using spot-welded aluminum alloy sandwich. The orthotropic core (longitudinal webs) sandwich was employed to give the high flexural rigidity needed in the longitudinal direction. This type of construction is generally most efficient when bending moments predominate in one direction, as in the solar panel application. The webbed-core and truss-core not only stabilize the facing sheets and provide shear stiffness, but they also carry a large share of the compressive load running parallel to the core elements. The fact that the longitudinal core material is effective in carrying compressive loads permits the use of a denser core than that employed in the honeycomb-core sandwich. In the honeycomb-core sandwich,the core material is not fully effective in carrying the loads in a cantilever beam, and an excess in core density can introduce serious weight penalties.

The high vacuum of space may present problems in the selection of both bearing materials and lubricants for the pivot points on solar panels. It is recognized that chemisorbed gases, including water vapor and oxide films, play an important role in friction. Once these are removed, localized "cold-welding" may result in seizing or galling of any metal-to-metal rubbing contact, providing the contact force is sufficiently high. Furthermore, lubrication of such moving or sliding contacts under high vacuum is very troublesome. It is known that the lubricating properties of graphite, for example, depend on the presence of a chemisorbed film of water vapor, and this is lost even at relatively low altitudes. Nevertheless, it is generally believed that hinged panel structures that are deployed within a few days of launch should experience little or no difficulty. Additional factors which tend to minimize the bearing problem are: (1) the movement will probably occur only once and in an unidirectional manner; (2) inertia considerations will dictate relatively slow motions; and (3) due to weightlessness, the bearings will be subjected to very light loads.

The power demands of most present-day scientific satellites are relatively small (less than 25 watts); however, power requirements can be expected to increase sharply into the multikilowatt range for sophisticated probes now in the planning stage and manned vehicles beginning with Apollo. It has been estimated that about 30,000 silicon-type solar cells would be required to make an array producing only 1 kilowatt of power. The array would cover about 100 square feet of panel surface, and the resulting solar panel would weigh at least 100 pounds. These rigid panel arrays would not only be complex and heavy, but they would be quite cumbersome to package. A possible solution to this problem is now being studied by NASA and their contractors[22], whereby large-area thin-film solar cells having high flexibility are under development. These cells will have low efficiency; however, they can be rolled up or folded into small volumes. Furthermore,

the evaporation process used to make these flexible cells is simple and very adaptable to large areas. A 1-kilowatt thin-film cadmium sulphide cell array of the flexible type is estimated to weigh about 70 pounds. Therefore, this appears to be an important area in which flexible concepts can be expected to replace semi-rigid concepts in the near future.

PETAL-TYPE SOLAR COLLECTORS OR ANTENNAS

Solar collectors with diameters ranging between 15 and 50 feet are now in both the planning and development stages. Most of these designs are to be adaptable to either the present scientific satellites or the early manned vehicles beginning with Apollo. The petal-type collectors present structural problems that are similar in nature to those of the solar panel; however, many more panels and pivots are involved, which complicates the precise alignment of reflective elements upon deployment. In these collectors, the alignment and dimensional tolerances must be carefully controlled, since slight surface irregularities can result in significant drops in reflective efficiency. The materials considerations that will be involved in the development of solar collector concepts include

(1) Bearing and lubrication materials subjected to space environment.

(2) Dimensional stability of the collector materials during a multitude of thermal cycles between high and low temperatures.

(3) Degradation of the quality of the reflective surface due to meteoritic dust, meteoroid penetration, sputtering, and related factors.

(4) Degradation of erected structural components under space environment.

A typical petal-type solar collector is displayed in Figure 13. As indicated by this drawing, most solar collectors will be employed in a space power conversion system in which the solar energy is focused onto a boiler unit. The space power plant shown here is that of a 15-foot diameter, 1.5-kilowatt concept patterned after an existing unit, designed and fabricated by Ryan Aeronautical Company. The mirror petals are of very lightweight, resistance welded construction using aluminum alloys. The stiffening used on these petals by Ryan is that of an open truss-work giving localized stiffening immediately behind the panels and an overall adjustable lattice-work that forms into a rigid system of Warren trusses to give a paraboloid shape upon full deployment.

A large 3?-foot diameter sunflower collector is now being constructed by Thompson Ramo Wooldridge, Inc. It is estimated that their overall collector system will weigh 700 pounds, however, the structural weight of the parabolic dish will be only 143 pounds. The system is designed to deliver 3 kilowatts of power for a period of one year. The large paraboloid can be folded during launch by rotating and skewing each petal about an individual hinge line so that the petals overlap and form a cylindrical package 10 feet in diameter. Upon deployment, the configuration is spread out in an umbrella fashion. The rigid petals of this design consist of

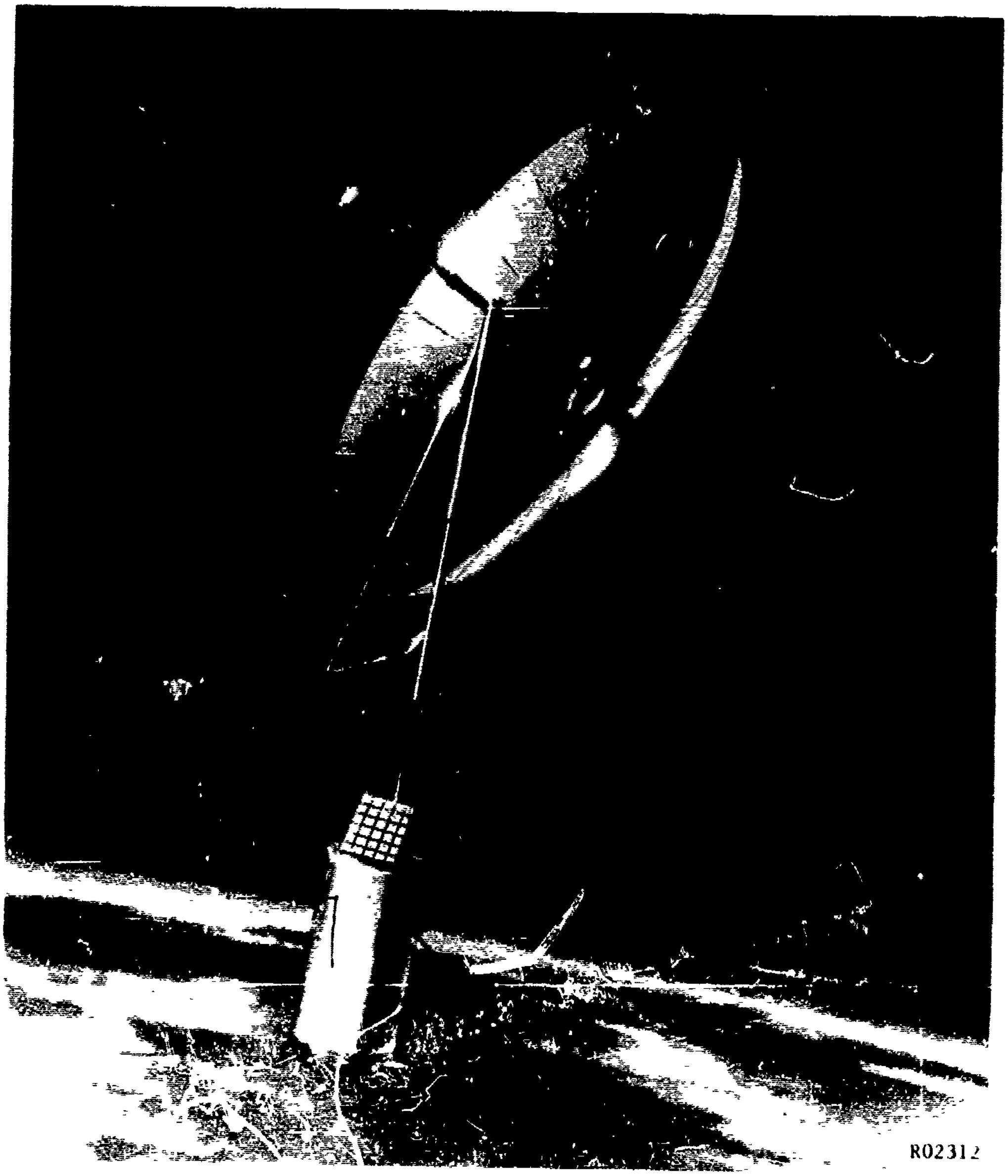

FIGURE 13. SEMI-RIGID SOLAR COLLECTOR

3-mil aluminum skin bonded to a honeycomb core of 1/2 inch depth. The concave surface of each petal is formed to hold the shape of a segment of a paraboloid. A centrally mounted tubular framework holds a boiler system at the focus point so as to receive the concentrated solar energy reflected from the highly polished surfaces.

A number of collectors of a similar nature are being developed by other contractors. However, each of these contractors are using a slightly different scheme for petal construction and deployment. General Electric Company has developed petal-type collectors similar to the above in which conventional sandwich materials are used for the petal elements. Electro-Optical Systems, Inc., on the other hand, is fabricating petals for their design from thin nickel sheet by the electroforming process. An overcoating of aluminum is then applied to this design to give the desired reflectivity. The Allison Division of General Motors is experimenting with a 4-foot diameter reflector of a radically different type. They are using the Fresnel concept, which resembles a serrated flat plate that can be easily folded into a compact package. After deployment, the serrations impart rigidity into the flat disk reflector. The unit is made by depositing nickel films onto a master, which in turn is coated by a reflecting surface of aluminum.

MANNED-TRAINER AND LABORATORY SPACECRAFT

The Mercury, Gemini, X-15, Dyna-Soar, and Apollo programs have already been initiated to identify and determine practical solutions to the problems of manned space flight. The Mercury program is completed, and the X-15 program is now in the advanced phases of flight research. Both of these programs have provided valuable data for future manned space flights. The Gemini, Dyna-Soar, and Apollo programs are at various stages of their funded programs, but actual flights appear to be some months or years away.

Each of the above programs have been planned so as to obtain definitive information relative to man's ability to cope with new problems in a strange environment, as well as to test the structural integrity of the craft involved. Although many of the anticipated conditions may be simulated in terrestrial laboratories, it has been found desirable to establish how well man and structure can perform over extended periods of weightlessness and amidst the simultaneous onslaught of the total space environment. This is especially important in considering any duty cycle or single task that may be required of an astronaut in the more advanced, expandable, space systems. To supply answers and to avert what may prove to be impossible assignments with regard to manned expandable spacecraft, additional programs will be required to provide information prior to the final design phase of any major space-station venture. These intermediate programs will require the use of "trainer vehicles" or "flight test craft" whose designs are based to a major extent on existing capability and state-of-the-art fabrication techniques, thereby meeting the earliest possible launch date.

A number of trainer-craft have been proposed by industry, and a typical trainer-concept under study by Aeronutronic[23] is shown in Figure 14. Here, a trainer-module (or mission-module) adapted to the Mercury capsule is illustrated; however, a similar configuration can be envisioned for the two-man

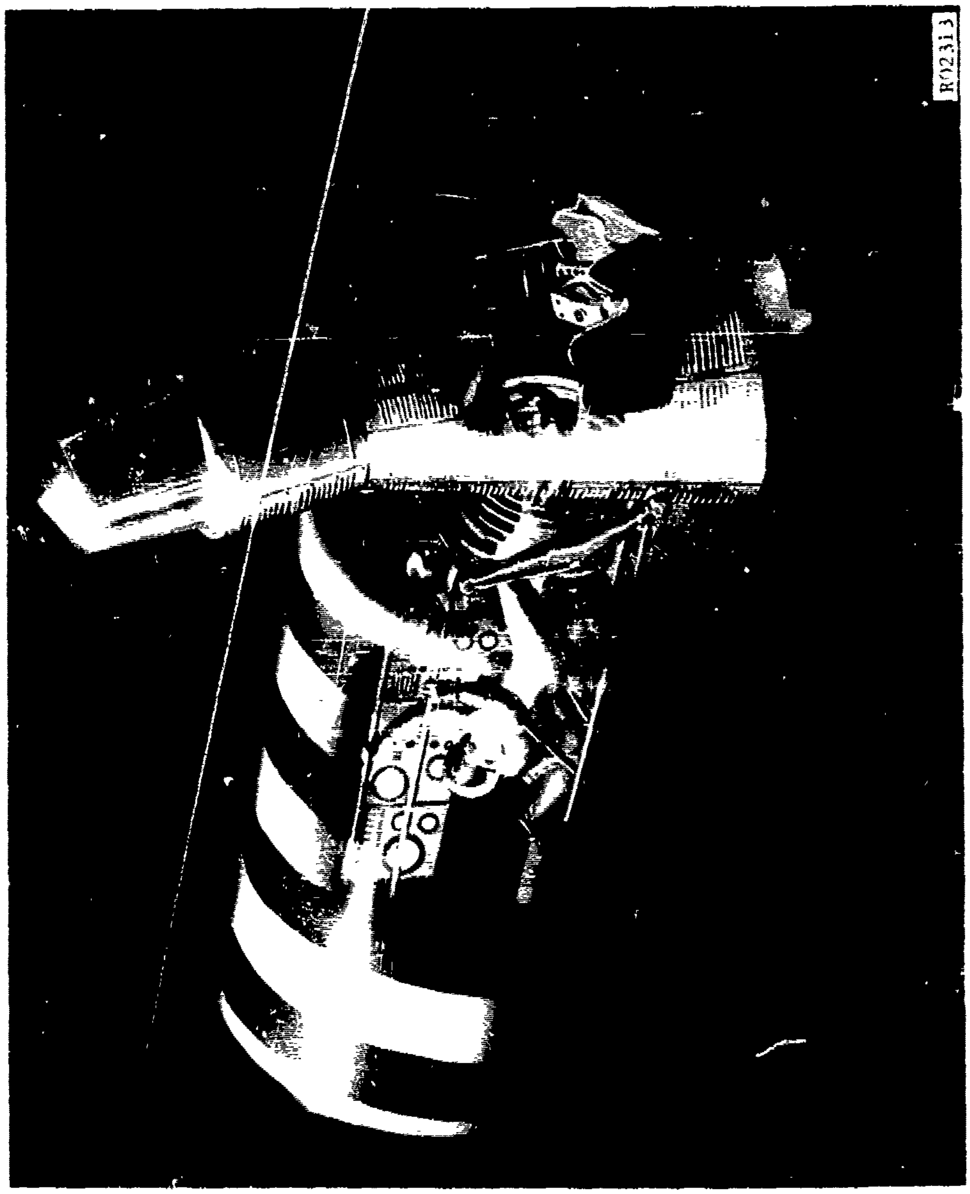

FIGURE 14. SPACE TRAINER - ORBITAL CONFIGURATION

Gemini capsule. The orbital orientation of the craft is shown in which a sealed passage is formed between the Mercury hatch and the module. This orientation is achieved by a relative rotation of the craft from the initial tandem arrangement employed during boost. Thus, it would be possible for an astronaut to enter the module and undertake a predetermined set of tasks related to an experimental program. As indicated in Figure 15, it is envisioned that the trainer-module will be adaptable to the performance of advanced expandable structure studies whereby the mechanics of deploying, sealing, and rigidizing structures can be observed and evaluated. Furthermore, for those experiments and structural evaluations requiring measurements over extended periods of time, both the expanded structure and mission module could be left in orbit. In this case, the astronaut would return to the re-entry capsule, disengage, and return to earth using re-entry techniques already developed. Structural evaluations and the retrieval of structural and material specimens from the orbiting craft would then be made at a later date by a second Mercury or Gemini capsule. For such maneuvers, the orbiting modules would serve as targets in trainer exercises involving rendezvous and docking procedures.

A much more sophisticated structures/materials space laboratory adapted to an advanced three-man re-entry capsule is illustrated in Figure 16. Laboratories and/or work shops similar in nature to this configuration may be envisioned for in-orbit inspection and component checkout during the assembly of large space stations or interplanetary craft. The vehicle shown in the figure is performing studies related to the merits of a semi-rigid concept which employs diametral expansion. These evaluations can be visualized to include the mechanics of deploying a new concept such as the "shingle" design shown here, the merits of new sealing schemes, the attenuation ability of structural arrangements or shielding methods, and possibly, the evaluation of composites for "self-sealing" penetrations by meteoroids.

TELESCOPING SPACECRAFT

The telescoping concept for deploying semi-rigid spacecraft is adaptable to a variety of module cross-sectional shapes (rectangular, hexagonal, cylindrical, etc.); however, in the application of this concept the cylindrical construction has received preference. Undoubtedly, this preference stems from the fact that the cylindrical module, for a given length and internal volume, presents a minimum surface area to the hazardous space environment. Furthermore, the cylinder displays a number of other design advantages. These include: (1) an external shape readily adaptable to booster vehicles; (2) a surface of revolution easily machined, fabricated, and stiffened; (3) a surface free of irregularities which, if present, would complicate stress analysis; and (4) an efficient configuration having uniform moment of inertia.

The telescoping concept, regardless of the cross-sectional shape of the module, is adaptable to rigid sections of either metallic or reinforced plastic construction. Rigid construction using sandwich composites, filament wound plastics, etc., is not far removed from that now being employed in aircraft fuselage and missile tankage structures. Therefore, expandable concepts making use of rigid sections show promise for early successful utilization in deployable spacecraft.

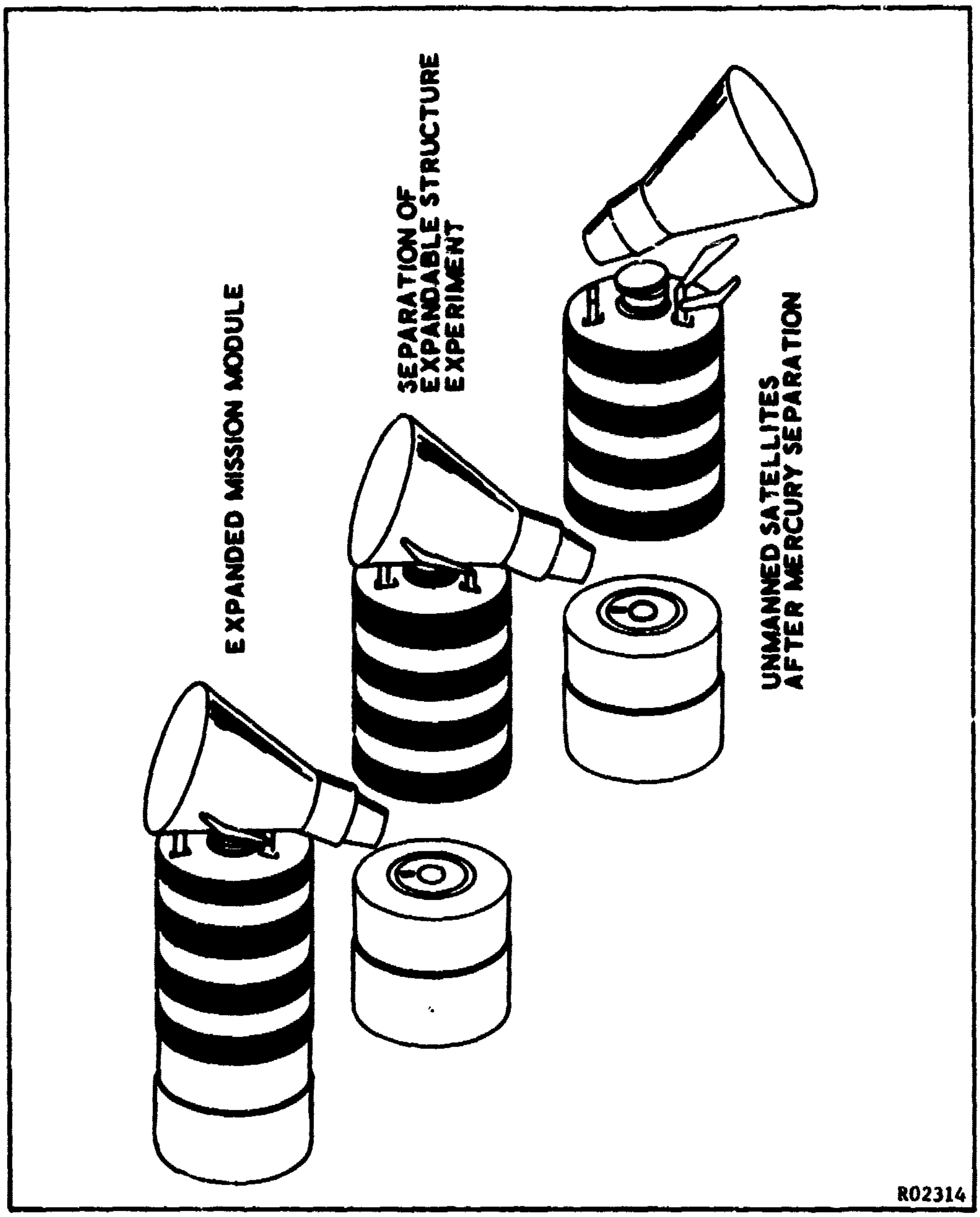

FIGURE 15. EXPERIMENTAL MISSION-WITH TRAINER CRAFT

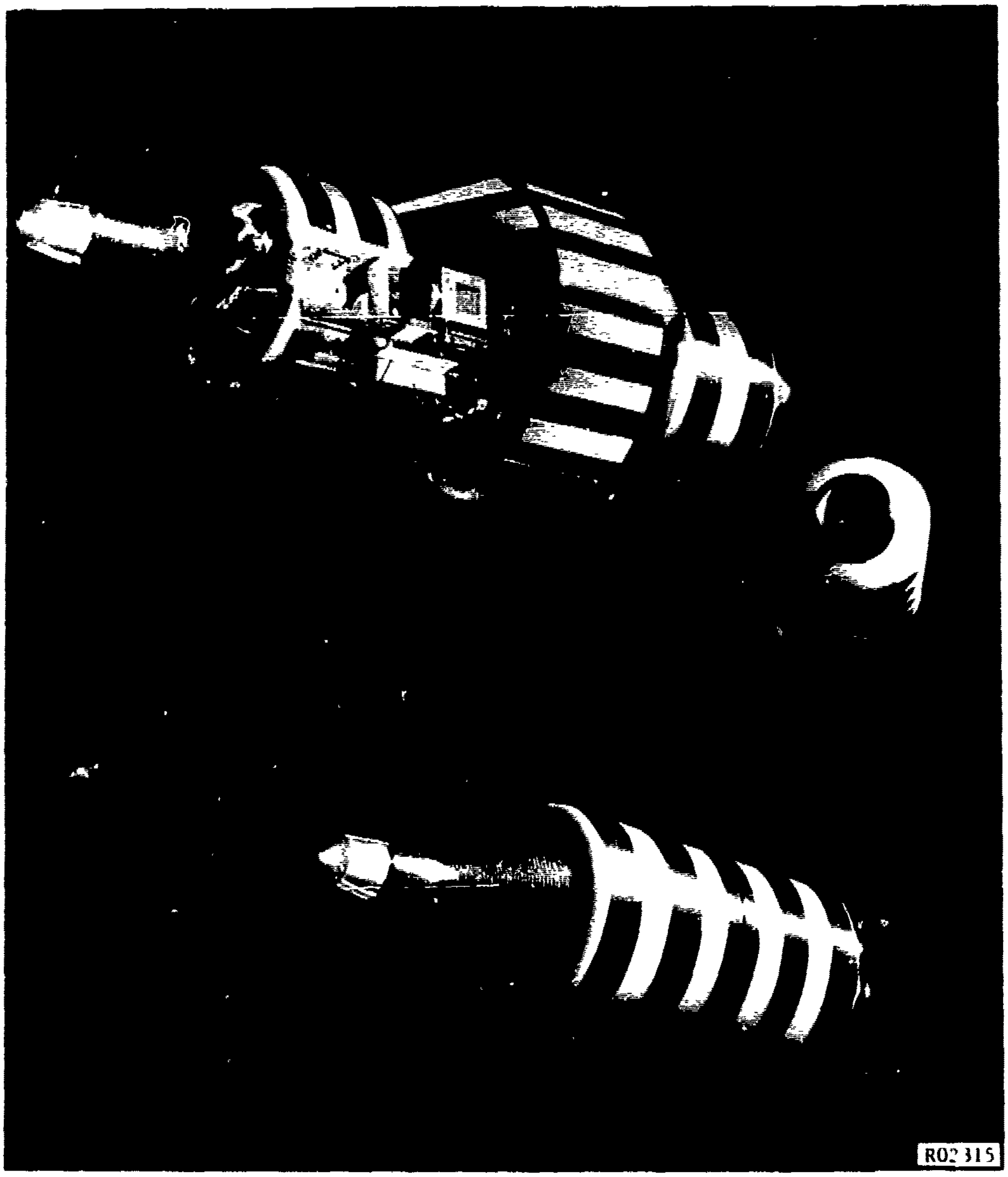

FIGURE 16. STRUCTURES AND MATERIALS SPACE LABORATORY OR WORK SHOP

A structural concept which has been under study at Aeronutronic for deploying a rather simple rotating space station is illustrated in Figure 17. This concept, referred to as the Rotating Dumbbell, is 100 feet in diameter, weighs 12,000 pounds, and accommodates 6 men. As indicated, the concept uses the telescoping feature throughout, including the passageway booms and the large-diameter laboratory modules. No flexing, folding, or hinging is required in the concept; therefore, the packaging and deployment will be in the simplest form possible for rigid sections. Furthermore, the simple process of packaging is adaptable to the recollapse of such spacecraft. This may well prove to be an important feature during an intense solar flare or meteoroid storm. That is, all available material could be used to hastily provide a multiwall storm cellar for protection against such occurrences which could be catastrophic.

The deployment of the telescoping concept is amenable to erection by gas pressure and/or mechanical means. Gas leakage during pressure erection can be held to a very low level by relatively inefficient seals, since deployment times and erection pressures will be very small. After full deployment, highly efficient, positive seals would be engaged at the overlap of each module so as to provide the necessary level of cabin sealing for extended mission times. The rigid-wall cross sections displayed in Figure 11 are applicable to the telescoping modules; however, additional rigidity in the form ring stiffeners may be required at the mating ends of each module. The requirement for load transfer across the overlap joints and the need for minimum distortion at the annular seals governs the design of these stiffening members.

A joint and wall concept that minimizes the need for heavy ring-stiffener designs is shown in the fully deployed position in Figure 18. This concept may be considered typical of that required of the passageway booms in the Rotating Dumbbell. Here, the construction is primarily chosen to satisfy requirements for thermal control and meteoroid shielding. Double-wall construction, without internal stiffening, is used with an aluminized Mylar superinsulation deployed (by unfolding) between the walls during vehicle expansion. The inner wall provides the pressure boundary for the vehicle interior as well as the structural integrity. As shown in the figure, the outer shell functions both as a meteoroid shield or bumper and a thermal radiator. The radiator tubes shown are attached to a glycol heat exchanger system and are an integral part of the outer shell.

The Martin Company (Denver) is now studying the manufacturing methods and processes required in the development of telescoping space structures. This work is being done under contract with ASD* and several interim reports[24,25] are available. As an end-item to this program, a typical full-scale telescoping spacecraft (zero-gravity type) will be manufactured and demonstrated. This will include deployment evaluations under simulated thermal environments and vacuum conditions.

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* Contract AF 33(657)-9733

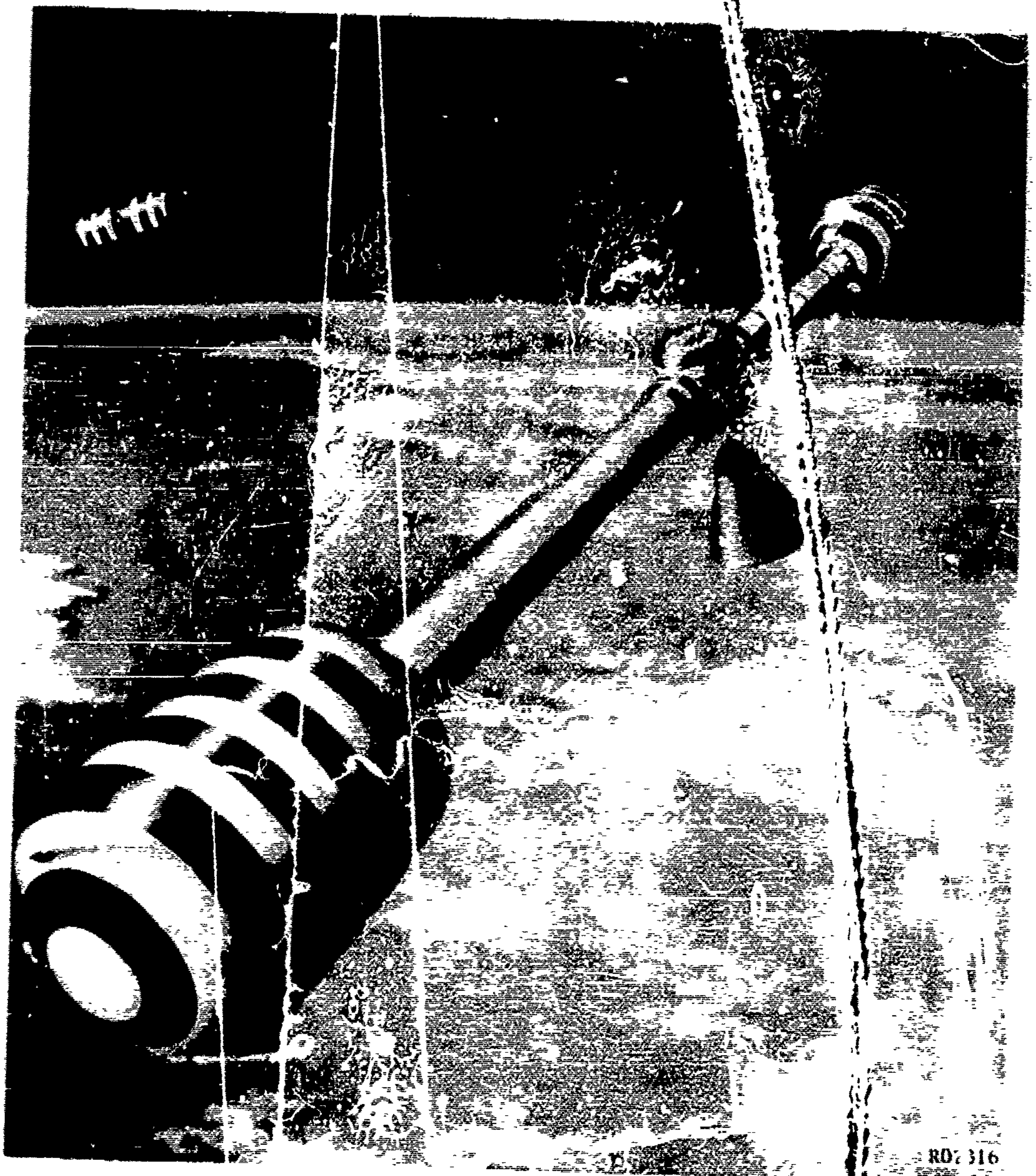

FIGURE 17. THE TELESCOPING DUMBBELL SPACE STATION

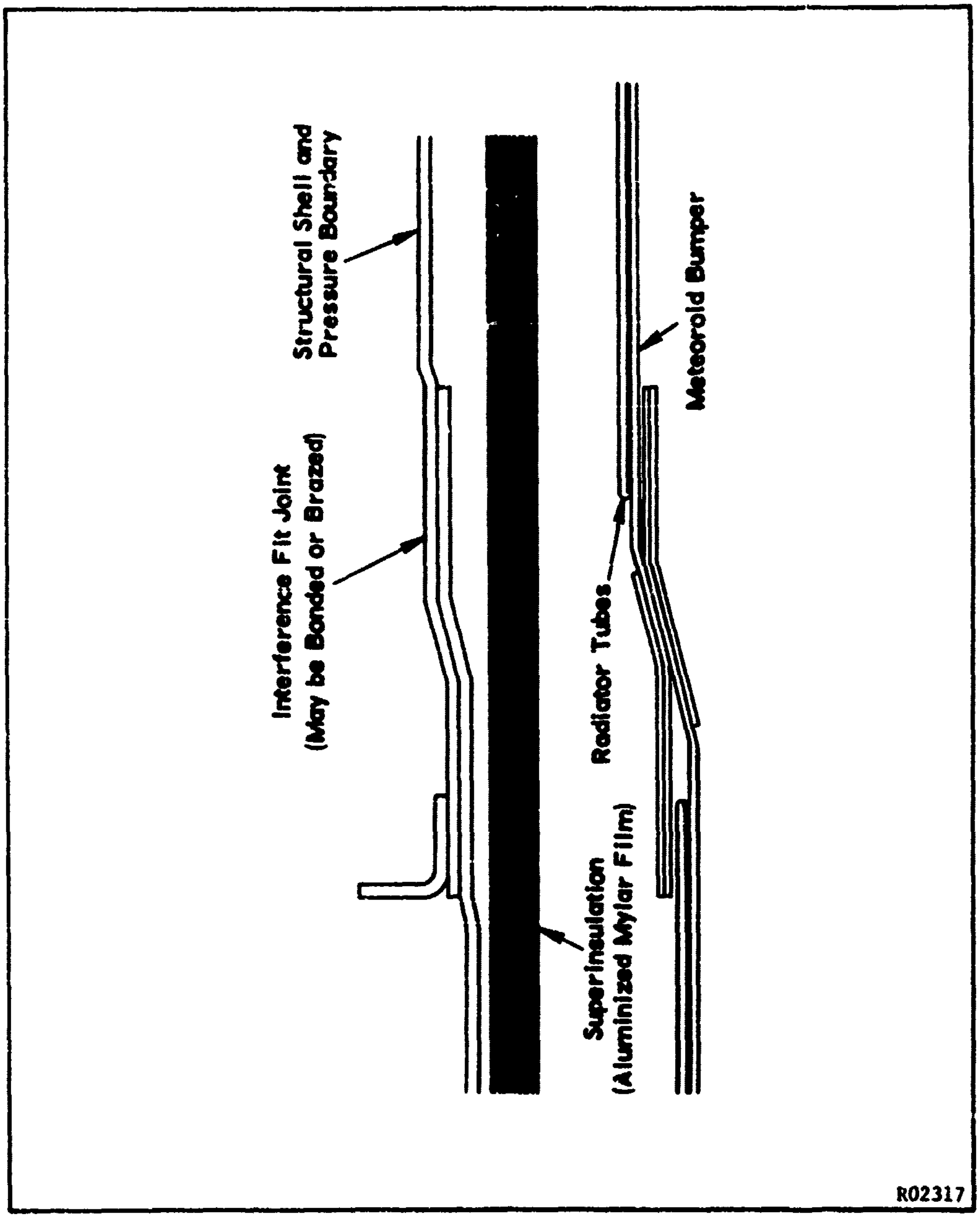

FIGURE 18. JOINT AND WALL CONCEPT FOR TELESCOPING SPACECRAFT

THE SELF-ERECTING SPACE STATION

Next to the inflatable satellites represented by the Echo balloon series, the use of expandable concepts has been more closely associated with the deployment of large rotating space stations than any other configuration. The large size of these vehicles obviously dictates the use of expandable structures of either the self-erecting type or that of assembly-in-orbit construction.

Designs for these large structural systems will depend heavily on how well technological developments on expandable structures accrue in the next year or two. That is, the approval of this next major step in the national space program is not only awaiting results from the preceding manned and unmanned space programs, but such approval is also awaiting clearer definition of missions and the feasibility of new structural concepts.

During the past several years, NASA has supported a number of study programs directed toward establishing the feasibility of designing a reliable manned space station. A summary of many of these programs has been made available recently.[26] In this summary, emphasis has been placed on the use of semi-rigid structures for space-station applications. One of the more comprehensive configuration studies was that performed by Space Information Systems Division of North American Aviation, Inc., in which the feasibility of deploying a number of semi-rigid, self-erecting space stations was investigated.

Originally, the self-erecting, manned spacecraft adopted for study was similar to that shown in Figure 19. Basically, this concept consisted of six rigid cylindrical modules joined by inflatable circumferential passageways and arranged, as shown, into a 100-foot-diameter hexagonal torus. A central hub was connected to this ring by means of three radial passageways of inflatable structure to allow access from hub to ring. At the hub, facilities were incorporated for docking two manned re-entry vehicles of the Apollo class. The configuration was to be sized so as to fold into a compact payload in which the rigid modules were individually rotated and tightly clustered into a large cylindrical package capable of being launched by a single Saturn-class booster. Deployment of the space station was to be performed by a combination of air pressurization and mechanically actuated struts connecting the hub and mid-point of each rigid module.

This self-erecting station was employed to illustrate most of the problems encountered in deploying complex space stations; however, as the research studies progressed, a number of more promising concepts were uncovered. Nevertheless, each of these newer concepts retained the "spoke" or "wheel" configuration, since it was found desirable to keep the mass of the station essentially in the plane of rotation. In the study by Space and Information Systems Division, it was found advisable to eliminate the requirement for exposed flexible fabric. Therefore, the inflatable passageways were replaced by telescoping spokes which could be retracted to one-half the deployed length for packaging. The flexible elbows between rigid modules were replaced with a system of compound hinges that enabled the modules to

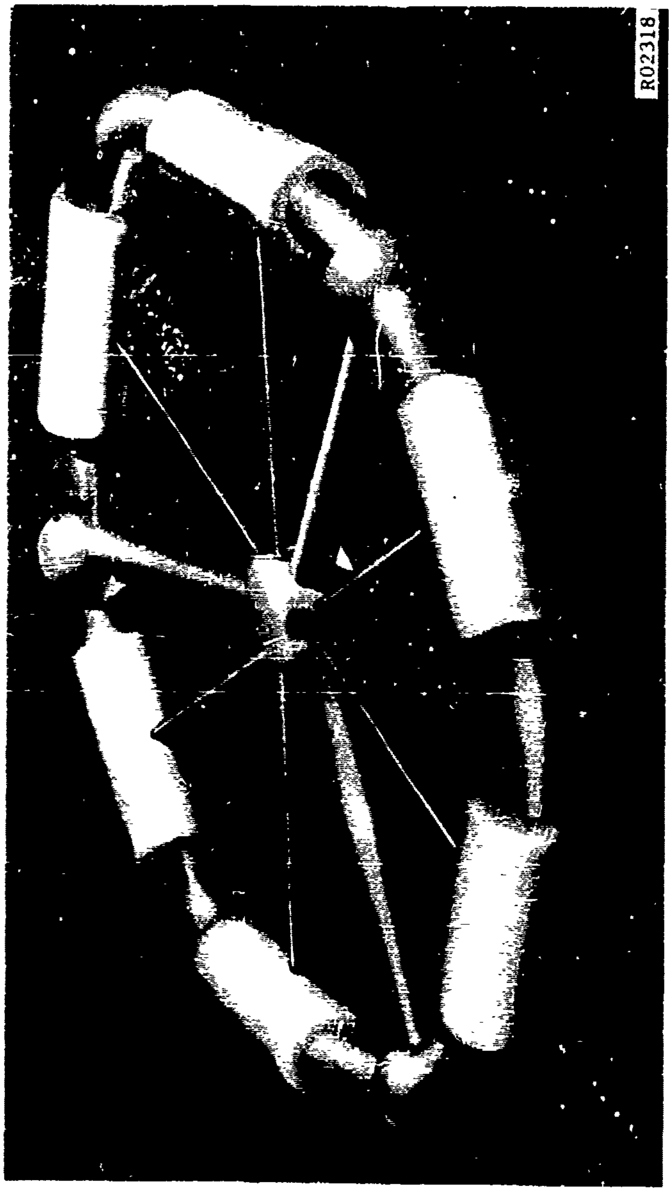

FIGURE 19. THE SELF-ERECTING SPACE STATION

rotate into the stowed position. This arrangement precluded deployment by pressurization; therefore, a series of mechanical screw-jack actuators were located at the joints. At full deployment, an airlock was engaged between adjacent modules to enable crew passage between these peripheral compartments.

In addition to the overall configuration studies described above, considerable effort has been expended in the area of structural wall requirements and the related areas of materials and fabrication techniques. Much of this work is also included in Reference 26. The conclusions reached in these programs emphasize the uncertainties that exist in the structural design of spacecraft due to the limited knowledge of the space environment. That is, the structural design and related weight of the space station is determined by the protection required against an environment that is very poorly defined. The wall cross section labeled (A) in Figure 11 is typical of that proposed from the results of these studies.

THE ASSEMBLY-IN-SPACE VEHICLES

The in-orbit assembly of a large manned space station or interplanetary vehicle will, without question, be the culmination of man's most complex engineering effort. This effort, considering our present limited knowledge of space and its effects, not to mention booster capability, must then be placed at the extreme of the time-scale shown earlier in Figure 1. Nevertheless, a number of basic design concepts have already been studied in considerable detail by industry. Typical of these concepts is that proposed for an advanced modular design[27] by Lockheed's Missile and Space Division. A simplified sketch of this configuration was shown in Figure 1. This assembled-in-space concept has the distinction of being granted the first space-station patent by the U.S. Patent Office. Two rigid module shapes comprise the essential elements of the Lockheed design. These consist of cylindrical sections 10 feet in diameter by 30 feet in length, and spheres 18 feet in diameter. Double-wall construction with longitudinal stiffening was employed for all cylindrical sections, and honeycomb sandwich was used for the spherical elements. The predominant material of construction was selected to be aluminum alloy. Assembly of the station, as proposed, would require the use of an "Astrotug" which is also under study by Lockheed. This smaller craft is a manned utility mover used for gathering up and coupling the several orbiting components so as to form the station. The main body of the assembled station is conceived to be about 94 feet wide and 108 feet long.

Another application of the assembly-in-space technique is that of constructing large interplanetary vehicles similar to that illustrated in Figure 20. This is a structural concept that was conceived by Aeronutronic for a six-man interplanetary spacecraft capable of an early (1970-72 time period) round trip expedition to the vicinity of Mars and/or Venus. Such a trip would take approximately one year to complete; therefore, the design of any concept of this type will be strongly influenced by the natural hazards of the space environment. The possibility of encountering a solar flare or meteoroid storm during this long mission will undoubtedly have a profound effect on the structural wall requirements and/or the arrangement of structural components so as to give a maximum level of shielding from these hazards. Furthermore, if man proves incapable of adapting himself to periods

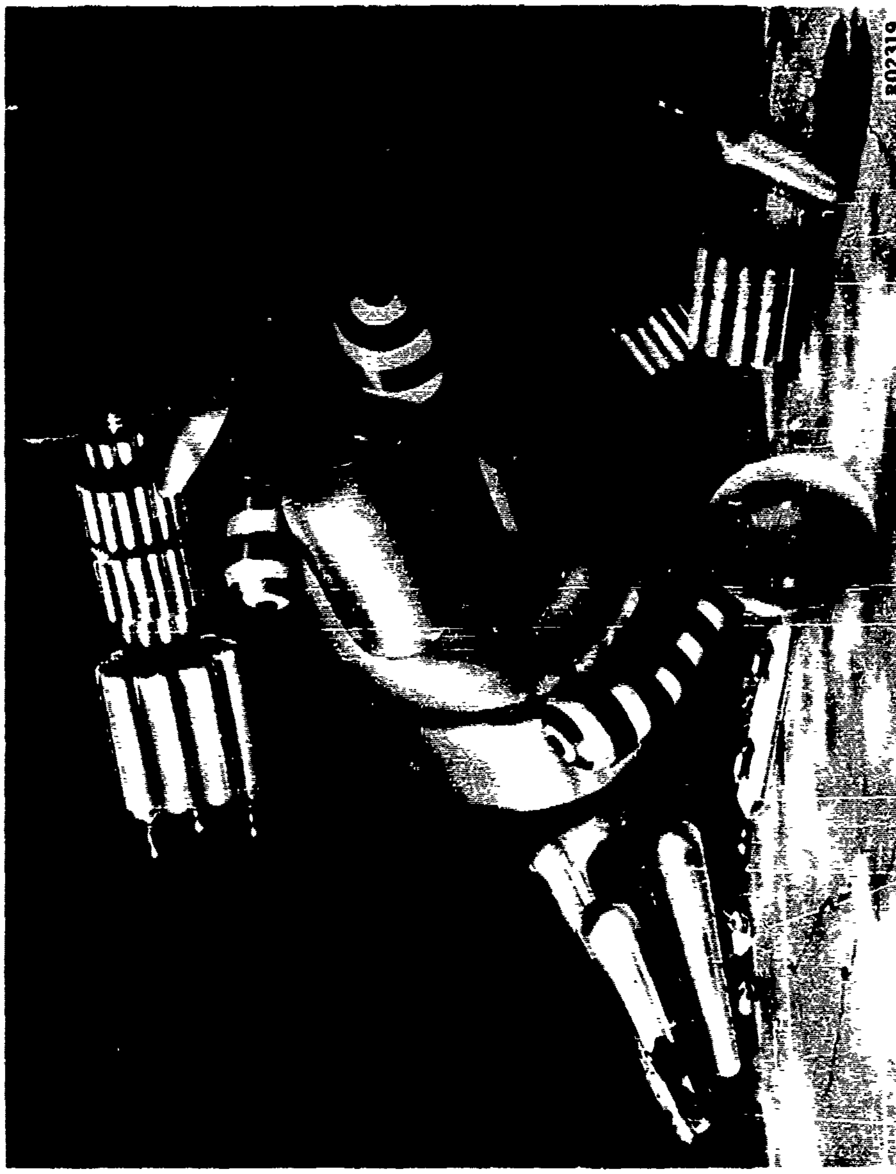

FIGURE 20. THE ASSEMBLY-IN-SPACE VEHICLE

of prolonged weightlessness, then it will be necessary to incorporate the rotational features of a space station to provide some degree of artificial gravity.

The configuration shown in Figure 20 reflects the stringent demands described above. The striped living modules are located at a radius of about 40 feet from the centerline of the vehicle, and the assembled vehicle is rotated to induce the desired "g" force. A storm-cellar (not clearly shown in the Figure) is located on the centerline and somewhat aft of the peripheral living modules. This orients the storm-cellar at the center of a pressure-vessel cluster, thus providing maximum security. Interconnecting tunnels would, of course, provide the necessary passageways.

As can be noted from the Figure, the greater mass of this interplanetary vehicle is composed of large tankage structure. This is due to the tremendous requirements for cryogenic propellant storage. Therefore, large weight savings can be realized through: (1) good tank design, (2) careful selection and installation of insulations, (3) proper vehicle orientation with respect to the sun, and (4) use of long-life, spectrally selected coatings on the external tank surfaces to provide a low α/ϵ ratio.*

The magnitude of the masses required in this vehicle clearly delineate the need for multiple launch capability. Presumably, either the Nova or C-5 vehicles would be required to meet the individual component-payload requirements. The cost of these systems dictates careful studies to maximize payloads and minimize the number of launches required.

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* Ratio of absorptivity to emissivity of coatings.

SUMMARY AND CONCLUDING REMARKS

Booster capability has been reviewed from the standpoint of evaluating the use of expandable concepts and the time-scale over which structural development will take place. The initial use of expandable structures for manned occupancy was shown to be operationally feasible at the present time by employing semirigid concepts in conjunction with a man-proven re-entry system. The orbiting of multiman, rotating space-stations, on the other hand, must await the operational use of the Saturn boosters. With the Advanced Saturn and Nova vehicles becoming operational in the late sixties, it was shown that assembly in orbit becomes feasible using multi-ton components.

For a manned operational mission, the paper has emphasized that the expandable spacecraft has certain requirements imposed on it which change its character considerably from that of merely a shape-changing and/or mechanical load-supporting device. It was indicated that the expandable living-modules must be compatible with the constraints imposed by the human requirements relating to internal thermal control, shielding against meteoroids and high-energy radiation, hermetic integrity, fail-safe design, and (possibly) artificial gravity forces during extended periods of time in space. Furthermore, it was shown that in the design of an expandable structure which meets all the requirements, a rather complex wall cross-section results. The wall complexity, in turn, directly affects the methods of packaging and the selection of schemes for deployment.

Aeronutronic has recently completed an investigation[28] for ASD under contract AF 33(616)-7775 which established both the structural and material requirements for expandable structures. This investigation involved the development of analysis and design techniques, parametric studies, materials evaluation, and typical configuration evaluation. Reference should be made to these results for a more detailed and comprehensive study of spacecraft design problems.

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AIRMAT STRUCTURES FOR SPACE AND LIMITED WAR APPLICATIONS

by J. T. Harris

Goodyear Aerospace Corporation

INTRODUCTION

Since the advent of international attention to the conquest of space, every technology has been scrutinized repeatedly in search of the most efficient system to perform each new assignment.

In this critical review, the field of expandable structures has received considerable attention. The passive communication satellite Echo I is the foremost working example of this technology which, in simple form, means the construction of a structure with materials which are foldable and can be deployed to their useful shape after orbit has been achieved.

Inflatable structures are a special case of this general class because their useful shape is maintained by internal pressurization. Primative forms of inflatable structures: spheres, cylinders and cones are bodies of revolution. The usefulness of these shapes is limited and means for extending it led successively from connecting cylinders together forming reasonably flat lobe type panels to completely woven panels called AIRMAT. *

This construction is an old weaving technology brought up-to-date by development of new machinery and utilization of new materials such as Nylon, Dacron and metallic yarns.

DISCUSSION

In the field of space structures the minimum weight, foldability and shape control of AIRMAT has made it a candidate material for many unique applications.

Space Antenna

Large lightweight space and lunar antennae are a prime requirement because of the long distances and limited booster payloads involved. One AIRMAT approach to this problem has been the consideration of a pill box

* T.M. Goodyear Aerospace Corporation

type antenna constructed of metallic AIRMAT. Figure 1 shows a scale model prototype currently under evaluation. It is a three inch section of a parabolic cylinder enclosed by two parallel planes consisting of silver coated stainless steel woven mesh AIRMAT with the focal point located at the center and edge of the aperature. It is center fed by an open ended waveguide. The spacing between the parallel woven cloth sheets is held constant by silver coated stainless steel threads and is held rigid by pressurizing the enclosure with air.

The advantages of such antenna construction are minimum weight and packageability for launch into space.

Paraglider

Another example wherein the incorporation of AIRMAT into a component of an inflatable structure provided distinct advantages occurred in an inflatable boomed Paraglider. The intersection of the inflated cylindrical booms and the keel at the apex confronted the design engineer with a difficult situation. Figure 2 shows a typical solution wherein the foldability of the apex is maintained by reducing the keel diameter prior to affecting the attachment to the curved cylindrical apex. Figure 3 shows the same intersection problem solved by introducing an AIRMAT apex which is a smooth transition of the three cylinders into the junction. This solution contributed substantially to the overall stiffness of the apex as well as aerodynamic smoothness.

The Paraglider concept has been considered for a low wing loading re-entry application as well as atmospheric recovery devices with the primary difference being materials of construction. In the atmospheric recovery devices, conventional textile yarns, such as Dacron, nylon and fiberglass, are acceptable materials while re-entry applications will require metallic yarns, such as stainless steel or Rene' 41 to sustain the heating effects.

Weaving Technology

For many years textile fiber AIRMAT has been available in thicknesses limited to approximately three inches by mechanical limitations on the weaving equipment. In order to exploit the potential of this technology GAC developed an experimental loom capable of producing aircraft quality AIRMAT material in both textiles and metals in thicknesses up to twice that commercially available.

Based upon this demonstrated capability, the Air Force granted GAC a contract to develop and construct a weaving facility capable of producing AIRMAT in thicknesses up to eight feet and 20 feet wide and to incorporate provisions for contouring the cross section. Figure 4 shows a view of the facility and Figure 5 and 6 show schematically a technological "breakthrough" in the method for producing the drop yarns resulting from this contractual effort.

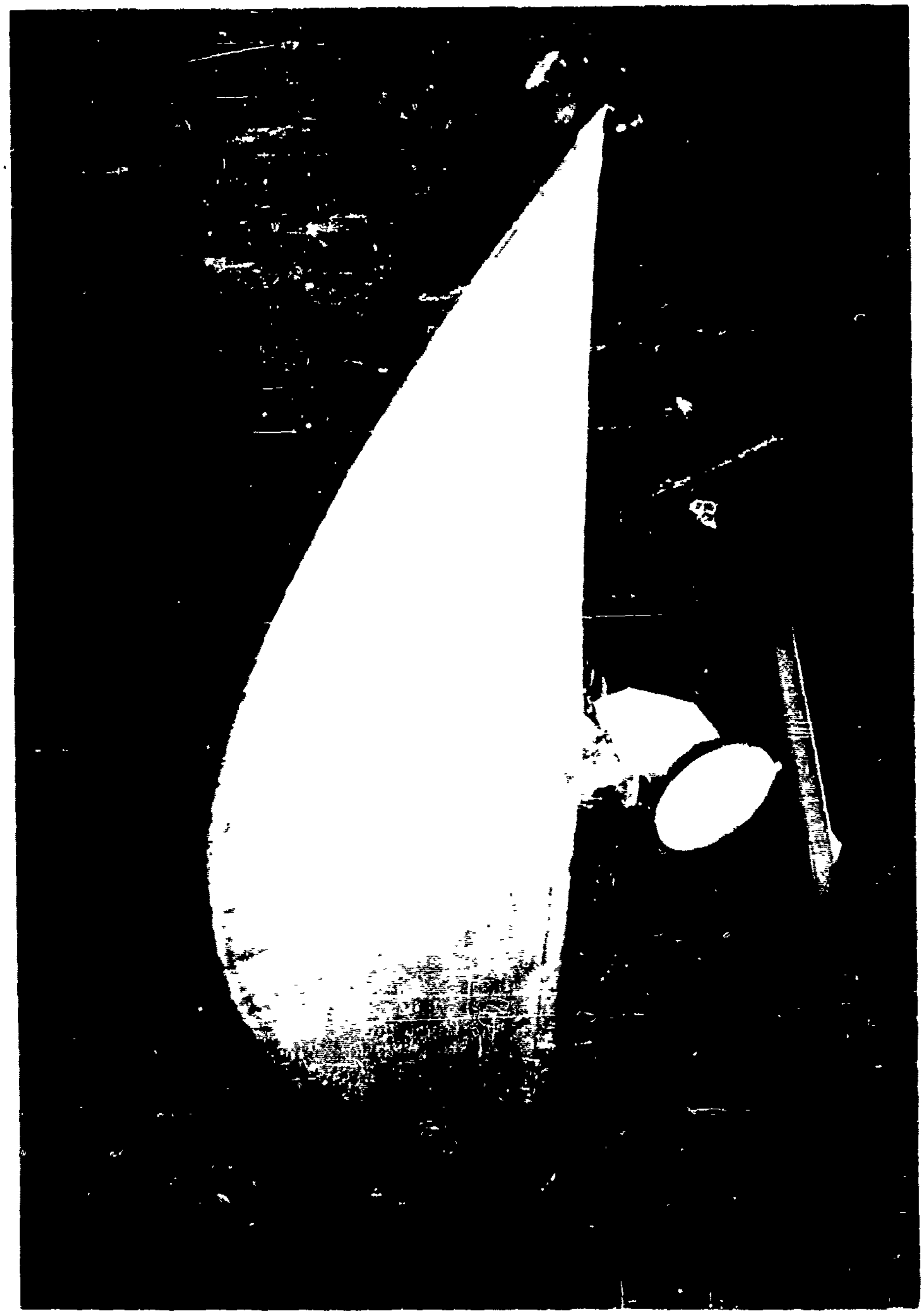

Figure 1, Airmat Structure Space Antenna

Figure 2. Paraglider Apex Tubular Intersection

Figure 3. Paraglider Apex Airmat Intersection


USAF 21-FT STRUCTURAL FABRIC LOOM
GOODYEAR
GOODYEAR AIRCRAFT CORPORATION

Figure 4

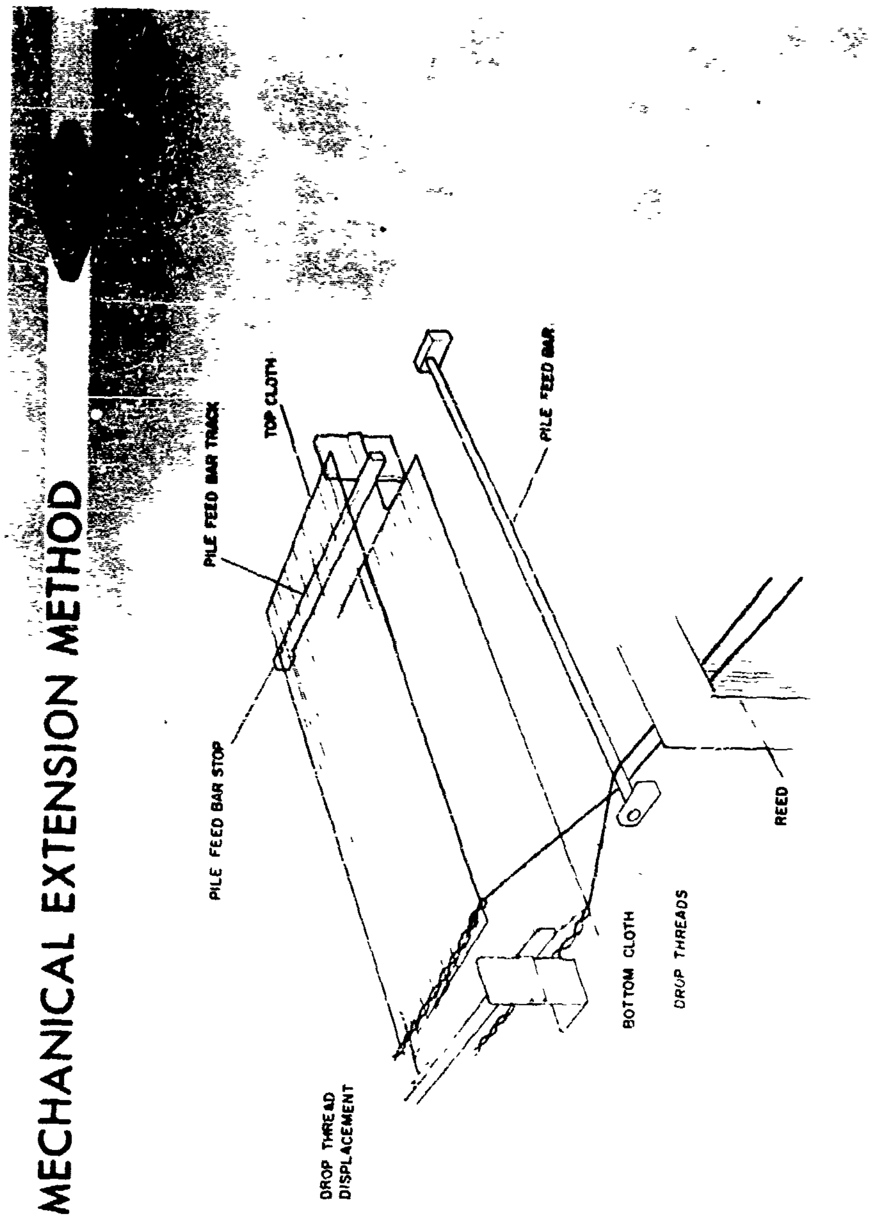

Figure 5. Airmat Mechanical Extension Method

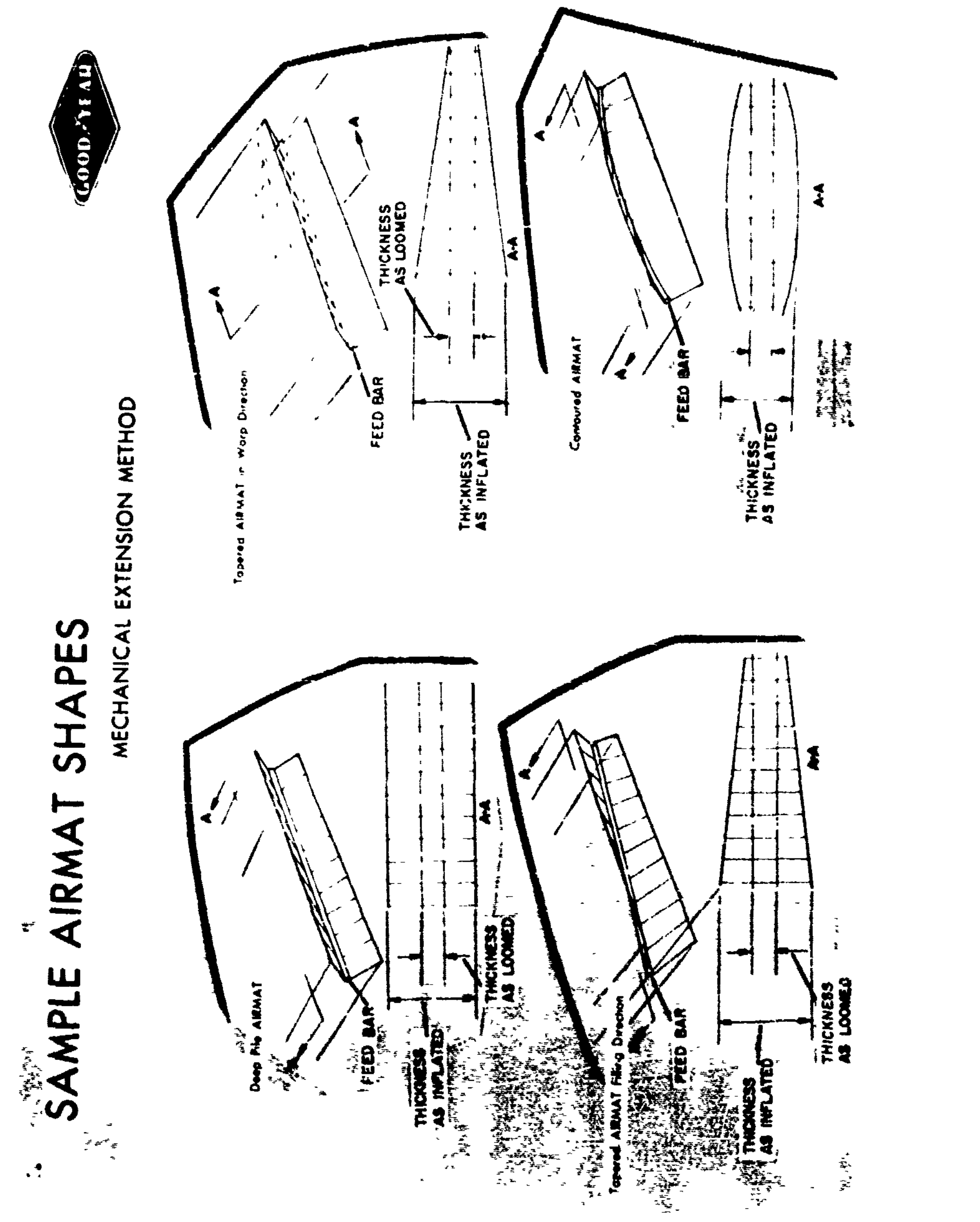

Figure 6, Sample Airmat Shapes

The basic principle involves entering the loom at the weaving sequence where the drop yarns leave their position in one cloth and transfer to the opposite cloth. At this time weaving is momentarily discontinued and the mechanical extension bar is introduced between the reed and the finished cloth edge. This bar is traversed horizontally between the upper and lower finished cloth to a preselected position determined by the desired length of drop yarn. At this time the cloth weaving is continued, thus locking this length permanently into the cloth.

Variations of mechanical extension bar traversing and bar shape can produce infinite variety of AIRMAT shapes from flat panels, wedges, airfoil sections with spanwise and chordwise taper to symmetrical concave or convex shells.

Rescue

Before the advent of space interest, AIRMAT was utilized in the manufacture of an all fabric airplane called the INFLATOPLANE* whose only metal parts were engine and landing wheel. The wing construction was a constant section of neoprene coated nylon NACA .0015 AIRMAT airfoil, the control surfaces and cockpit were flat panels of two inch thick nylon AIRMAT and the fuselage was a neoprene coated Dacron fabric cylindrical cone.

One military application of this unique airplane was recognized as a rescue vehicle which could be air dropped to stranded personnel. These personnel could unpackage the vehicle, inflate and effect their own rescue by flying out. Ten two-place versions of this airplane were delivered to the military for evaluation.

Ground Effect Jeep

Under the present complexion of ground military vehicle operation, terrain conditions are often such that additional floatation would enhance mobility enormously. A ground effects machine (GEM), a vehicle which operates on a cushion of air, has been considered for such operations but the power requirements and structural size have been prohibitive.

An interesting approach toward solution of this problem is currently under evaluation at GAC. It involves the adaptation of a flexible fabric skirt to a jeep frame and is shown in Figure 7. An inflatable support frame of AIRMAT and a fabric cylinder were built around the frame so that the structure could be deflated and collapsed when not in use and not interferewith the movement of the jeep on narrow roads or through small openings. When inflated, it increases the planform area considerably so that low air cushion pressures were obtained. The inflatable components also provide emergency flotation for the vehicle in the event of power loss over water. To provide air for the cushion, a fan was mounted in front and was driven from the fan pulley of the jeep engine so no auxiliary power plane was required.

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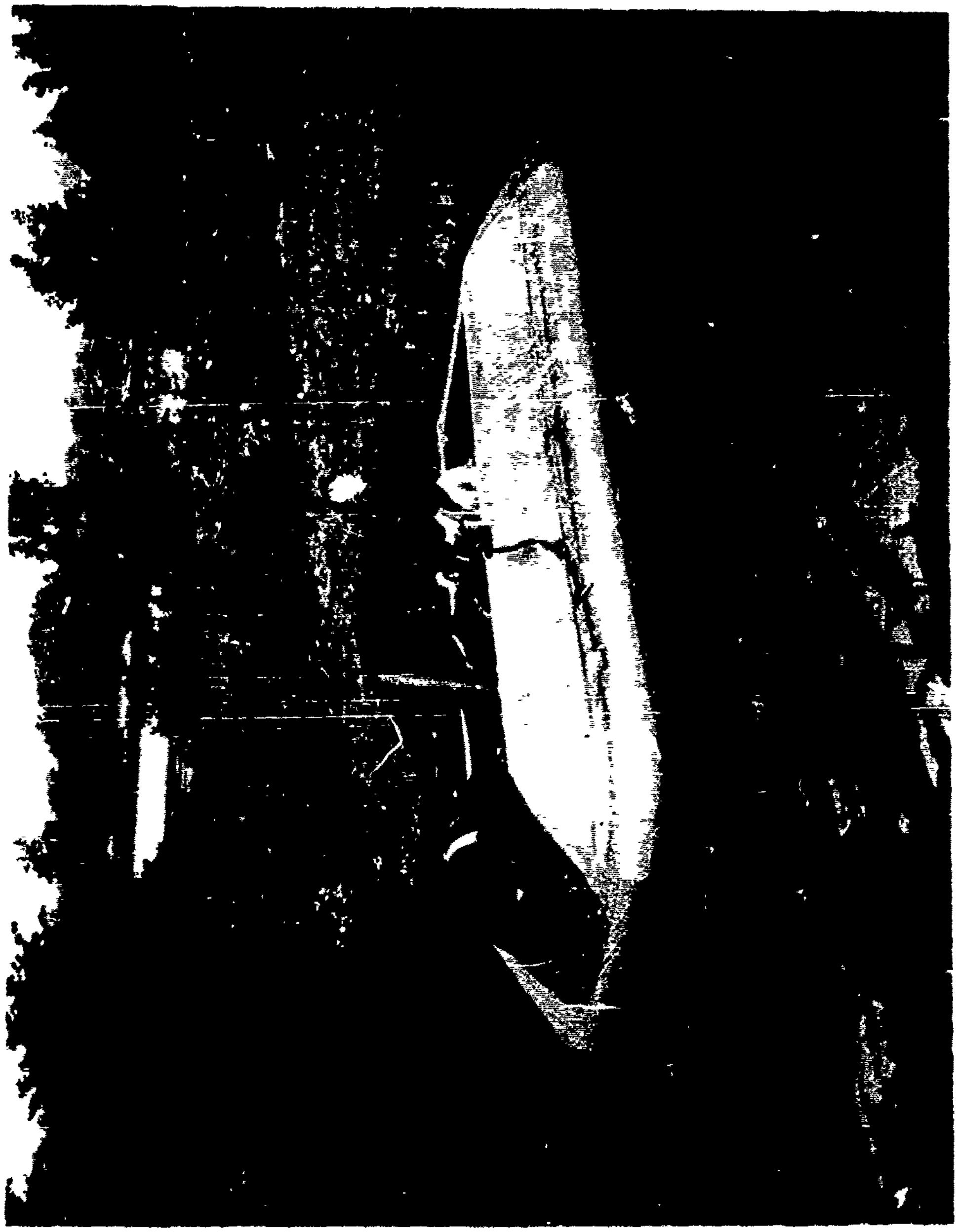

Figure 7. Airmat Shirt and Jeep

The air cushion was restrained by a flexible curtain operating about one inch off the ground. The low pressure and low operating height required only a portion of the jeep engine power.

The design objective was to remove most of the weight from the wheels so they would not sink into the soft ground and become mired while maintaining enough load on them to provide traction for propulsion and steering. Parametric test data in this area is being generated.

General Utility

Modern military operations are constantly requiring more electrical power. Even advanced outposts require more and more power for communications. Gasoline powered engine generator sets have been developed to a high degree and the UGP-12 represents lightweight compact unit. However, the noise developed in its operation is annoying to an operator in constant proximity as well as a hazard because of easy enemy detection in advanced locations. Rigid sound insulation would add considerably to the package bulk and weight and compromise and otherwise successful development.

A packageable AIRMAT box was constructed to house the unit, Figure 8. It was lined with flexible foam and foam lined fabric intake and exhaust sleeves were attached. The box when packaged occupied about the same volume as the engine generator set itself. In operation the AIRMAT was filled with water. This combination of foam and water demonstrated a marked reduction in external noise level over the entire audible range.

This was affected with a minimum weight and transported bulk by transporting only the AIRMAT container and utilizing water available at most sites.

CONCLUSION

The application of the AIRMAT technology to some space and limited warfare problems offers unique solutions that are impossible by any other means and expanded utilization of this material seems limited only by engineering imagination.

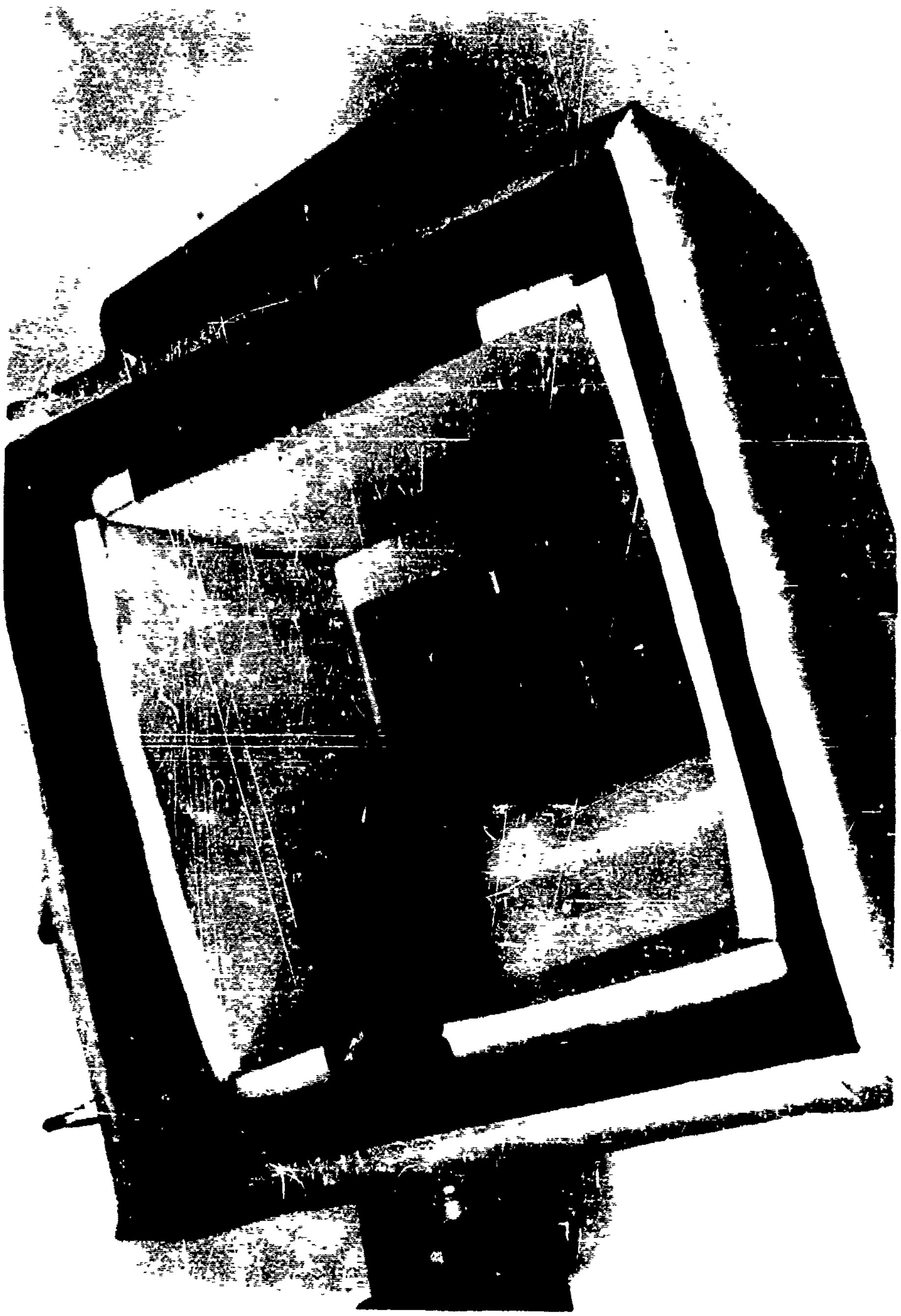

Figure 8, Engine Generator Airmat Silencing Box

FLEXIBLE FIBROUS MATERIALS AND COATINGS FOR EXPANDABLE RE-ENTRY SYSTEMS

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INTRODUCTION

The proposed use of expandable structures fabricated from flexible, compactable fibrous materials has attracted considerable attention from systems designers. The designs of numerous re-entry and lift drag vehicles and other space structures incorporate flexible, impermeable (through use of coatings), thermally stable high strength fibrous materials. Expandable structures formed from coated woven materials offer a number of distinct advantages over rigid structures, namely deployment control, compactibility, and lower weight. In applications where aerodynamic heating may occur, thermal stability can become a problem with expandable structures. Some of the proposed flight profiles will encounter stagnation temperatures in the range of 1500° - 2500°F. which eliminate the present day textile fibers unless unusually thick (and inflexible) coatings would be utilized, which would seriously affect compactibility and recoverability.

Two completely separate areas of research are involved which ultimately must result in the mating of the most suitable candidates from each to establish compatibility, compactibility and deployability. The ultimate composite structure must exhibit some degree of built in recovery or some little resistance to external recovery forces upon release of that force which creates deformation during compaction.

In the case of the reinforcement materials, superalloys appear to be the most suitable candidates for use as filamentary forms (Ref. 1) provided they can be made fine enough and can be successfully processed. The basis for utilizing multifilament metallic yarns are discussed as are the actual yarn and fabric design concepts studied. The feasibility of weaving ultra-fine metallic filaments in twisted form are also reviewed as are the problems encountered and the results obtained. The matrix (or coating) material, as in the case of the flexible reinforcement requires investigation into an entirely new concept of coating materials. The initial research was carried out by Goodyear Aerospace (Ref. 2.) and resulted in a coating which would withstand temperatures of 1200° - 1400°F., however upon cooling, shrinkage or bending creates sufficient cracks to effect permeability adversely. Further research to extend this new coating concept to higher temperature regimes is described. The improvements gained are shown using the flexible metallic fiber material attained in the fabric development phase.

DISCUSSION

I. Multifilament Yarn and Fabric Design

When considering the need for flexible fibrous materials to be used in expandable structures, the choice of the starting material is extremely limited. As shown (Ref. 1,3,4) by others the only substance with the thermal stability required falls in the metals class. Of primary interest have been the superalloy

type materials. High temperature alloys such as achieved with nickel-chromium systems (Chromels, Rene' 41, Evanohm, etc.) and the cobalt-nickel-chromium systems (Elgiloy) have the strength at temperature required to withstand re-entry environments. The major problems have been (a) ability to be drawn to ultra fine diameters, and (b) can these fibers be formed into flexible yarns, and subsequently be processed into efficient woven structures. The former is being adequately dealt with (Ref. 5,6,7) and would constitute a separate paper. The latter is to be discussed in this paper. For the purpose of this discussion, how the single ultra fine fibers were achieved shall not be considered, only that continuous fibers of 1.0, 0.7 and 0.5 mils diameter were obtainable.

The question naturally arises, why are such fine fibers required. To achieve the flexibility desired, low bending rigidity is implied. Metallic fibers, as a group, are 10 to 20 times more rigid than organic fibers of equal strength. This difficulty can be overcome by reducing the diameter of the metal fibers and increasing the number of fibers in the yarn bundle to retain the same tensile strength. The increase in flexibility in a single fiber has been found to be increased by a factor of 4, in reducing the diameter by 50% (Ref. 1). Much additional research has been conducted recently into the theoretical advantages of using ultra fine diameter fibers (Ref. 8). It has been shown that fabric flexibility, bending recovery and fold endurance are functions of fiber and yarn flexibility and bending recovery. These, however are not the only considerations. The fabric structure; yarn spacing, weave pattern, is also an important variable. As an example of this, the occurrence of yarn flattening during the weaving of a fabric results in lower fiber strain during fabric bending and thus greater flexibility, fold endurance and bending recovery. The amount of yarn flattening that occurs is dependent on the twist imposed on the yarns. Low twist results in more flattening than high twist. Similarly the construction, i.e., ends and picks per inch, and the weave pattern effects yarn flattening. Other factors which will effect fold endurance, bending recovery and flexibility are the amount of crimp in the yarns after weaving which can be varied by weaving tensions, and by using opposite twist in the warp and filling yarns to prevent nesting. Nesting restricts yarn movement and therefore decreases flexibility.

Since fabric porosity and coating adhesion are of considerable importance for re-entry applications, a blending of fibers per yarn, twist, yarn count, weave pattern, weaving tensions must be affected to provide a flexible material capable of fulfilling specific system requirements.

From these theoretical conclusions it can be concluded that a low twist multi-filament yarn composed of many ultra fine diameter fibers, and woven into a square weave, with medium float length, would achieve a structure with mechanical properties realistically close to those calculated for re-entry systems. With this in mind, the design and fabrication of woven structures were then considered. Of prime importance was the utilization of textile type twisting and weaving equipment. To thoroughly verify the theory concerning flexibility, fold endurance, wrinkle recovery and tear as discussed, three fiber diameters were chosen to be woven into a series of fabrics. The main concern was to achieve roughly equal yarn strength regardless of the fiber diameter used. To this end one mil wire was twisted into 25 filament bundles, 0.7 mil diameter wire into 49 filament bundles, and 0.5 mil fibers into 100 filament bundles. To achieve a torque free or balanced yarn, two twisting operations were conducted. For example, to obtain 100 filament yarn of 0.5 mil fibers, an initial twisting of 10 single filaments were accomplished. Ten of these yarns were then cabled

with opposite twist which resulted in a 100 filament balanced yarn.

The alloy used is Chromel A, a 80-20 nickel chromium alloy readily available in the desired diameters. It has room temperature properties very similar to the high temperature alloys now being considered for re-entry decelerators and expandable lift-drag vehicles. Another goal of this program was to establish techniques for processing these metal fibers on textile type equipment. Previous work on metal fibers has basically involved stranding devices such as used in forming cables and very limited capability (from a fabric design standpoint) looms.

The first step was to process the three metal fiber diameters through a textile twister. In this case a slight modification in the creeling system was made to reduce the drag on the single fibers in the first twisting operation. Further changes included elimination of the stop motion when processing 0.5 mil fibers. Table I shows the fiber and yarn properties while Figure I reveals the load elongation curves for the fibers. The only serious problem encountered in twisting was caused by the spool the metal filaments were delivered on, and damage that occurred (on 0.5 mil fibers) during winding on to the spool and the putting of too much weight of fiber on the spool. These were overcome by sanding and polishing the flanges and barrels of the spools and limiting the weight of fiber per spool to approximately .02 to .04 lbs.

The cabling operation created no problems although special consideration was given to the traveler used. The yarn was made into a 9-1/2 yard warp with sufficient ends to weave a 6 inch fabric. A silk type loom was used and the warp yarns drawn into eight (8) harnesses to afford weaving a wide range of fabrics. The filling yarn (of the same construction as the warp yarn) was wound on a conventional quill and woven at a rate of 110 picks/minute in a standard shuttle. A total of eight weave designs were used ranging from a plain weave, to 2x2 basket (the densest), to a 3/1 twill, to a 2/2 twill, to a combination 3/1 warp weave with a 1/3 filling weave alternating the metal filling with an HT-1 filling. The fabric properties achieved are shown in Table 2, however, only three fabrics are presented for each fiber diameter. (Fig. 2,3,4). As a basis of comparison a monofilament fabric was also evaluated. Tables 3 and 4, present data on wrinkle recovery - fold endurance, strength, tear and elongation. Fig. 5 shows the load elongation curves for 0.5 mil fiber from single filament through yarn and into fabrics.

This program emphatically demonstrated the feasibility of twisting metal fibers into yarns and weaving them into an assortment of fabrics. It can be seen that from monofilament to multifilament yarn the change in properties for

a. Tear - is up to 4 times higher

b. Fold endurance - is 20 to 100 times greater

c. Wrinkle recovery - is up to 39%

d. Permeability - is reduced to 0.01

e. Strength - is two to four times higher

To overcome the only problem area - twisting of the 0.5 mil fibers, a program is now being carried out to develop a twisting facility having the extremely fine

table 1 FIBER AND YARN PROPERTIES

Dia. Mils	Fiber & Yarn Construction	Strength (lbs)		Elongation (%)		Mod. of Elasticity (psi x 10^{-5})
		Yield	Rupture	Yield	Rupture	
0.5	Single Filament	0.026	0.029	0.73	8.52	22.6
	10/1/3.1Z tpi	0.259	0.284	0.89	6.43	20.5
	10/10/1/3.1S/3.1Z tpi	2.568	2.820	0.81	6.94	19.8
0.7	Single Filament	0.034	0.048	0.43	14.50	25.5
	7/1/3.1Z tpi	0.231	0.314	0.83	9.72	16.3
	7/7/1/2.7S/3.1Z tpi	1.724	2.299	0.83	10.33	17.8
1.0	Single Filament	0.104	0.112	0.71	11.51	23.5
	5/5/1/2.0S/3.1Z tpi	2.546	2.745	1.20	6.04	12.7

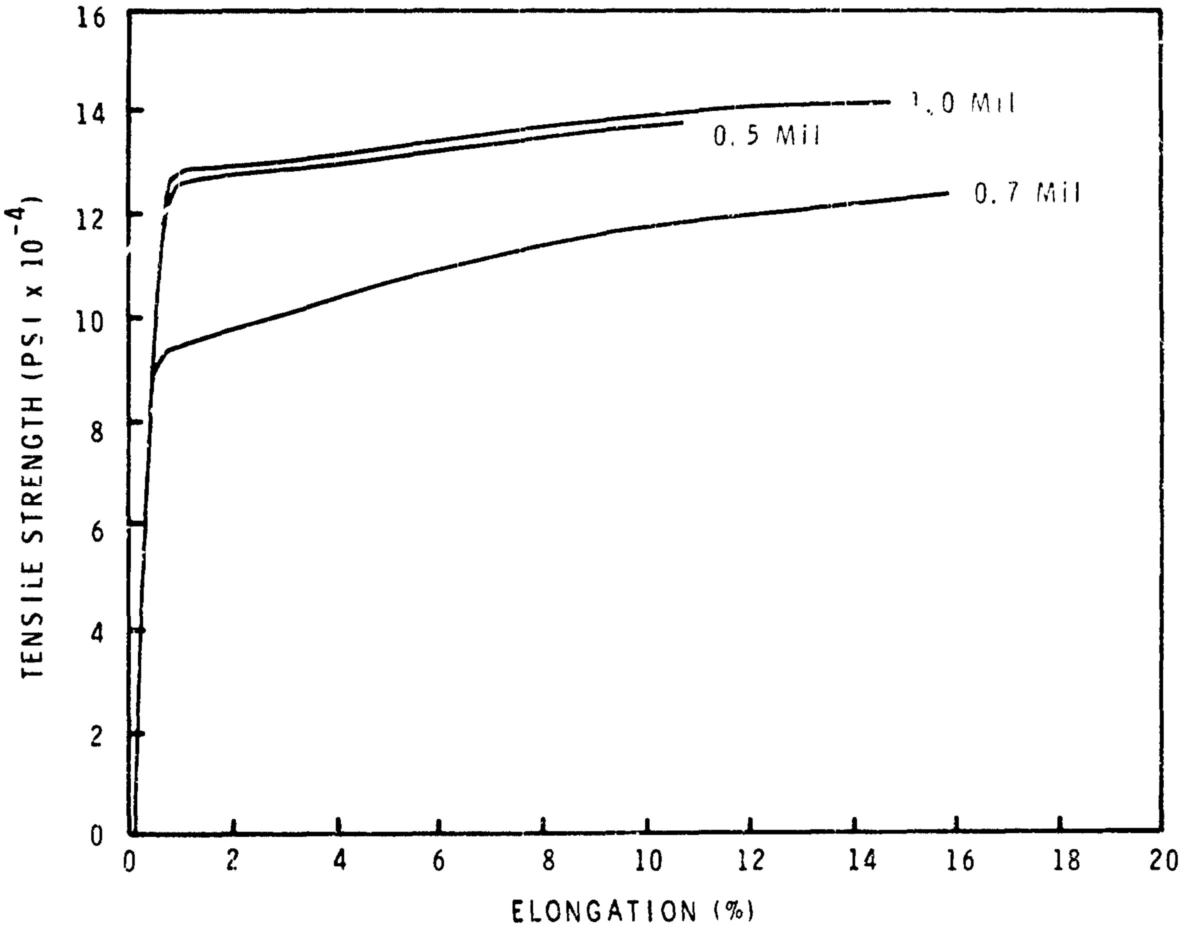

figure 1 TYPICAL METAL LOAD-ELONGATION CURVES

Table 2 FABRIC PROPERTIES

Fiber Dia. Mils	Material & Yarn	Weave Design	Ends per inch	Picks per Inch	Weight (oz/yd^2)	Thickness (inches)	Permeability (ft^3/min/ft^2)
0.5	Chromel A	Plain	61	61	16.1	0.0067	52-57
	10/10/1/3.1S/3.1Z	2 x 2 Basket	81	81	21.2	0.0072	18-21
		2/2 Twill	82	82	21.3	0.0073	11-14
0.7	Chromel A	Plain	81	75	19.0	0.0081	26-32
	7/7/1/2.7S/3.1Z	2 x 2 Basket	82	119	23.9	0.0082	8-11
		2/2 Twill	82	102	21.5	0.0074	13
1.0	Chromel A	Plain	60	61	15.1	0.0073	131-137
		2 x 2 Basket	81	81	20.1	0.0081	47-57
		2/2 Twill	81	81	20.1	0.0083	30
1.5	Rene' 41 Monofilament	Plain	200	200	4.7	0.0035	1110
3.0	304SS Monofilament	2/2 Twill	157	157	13.3	0.0073	888

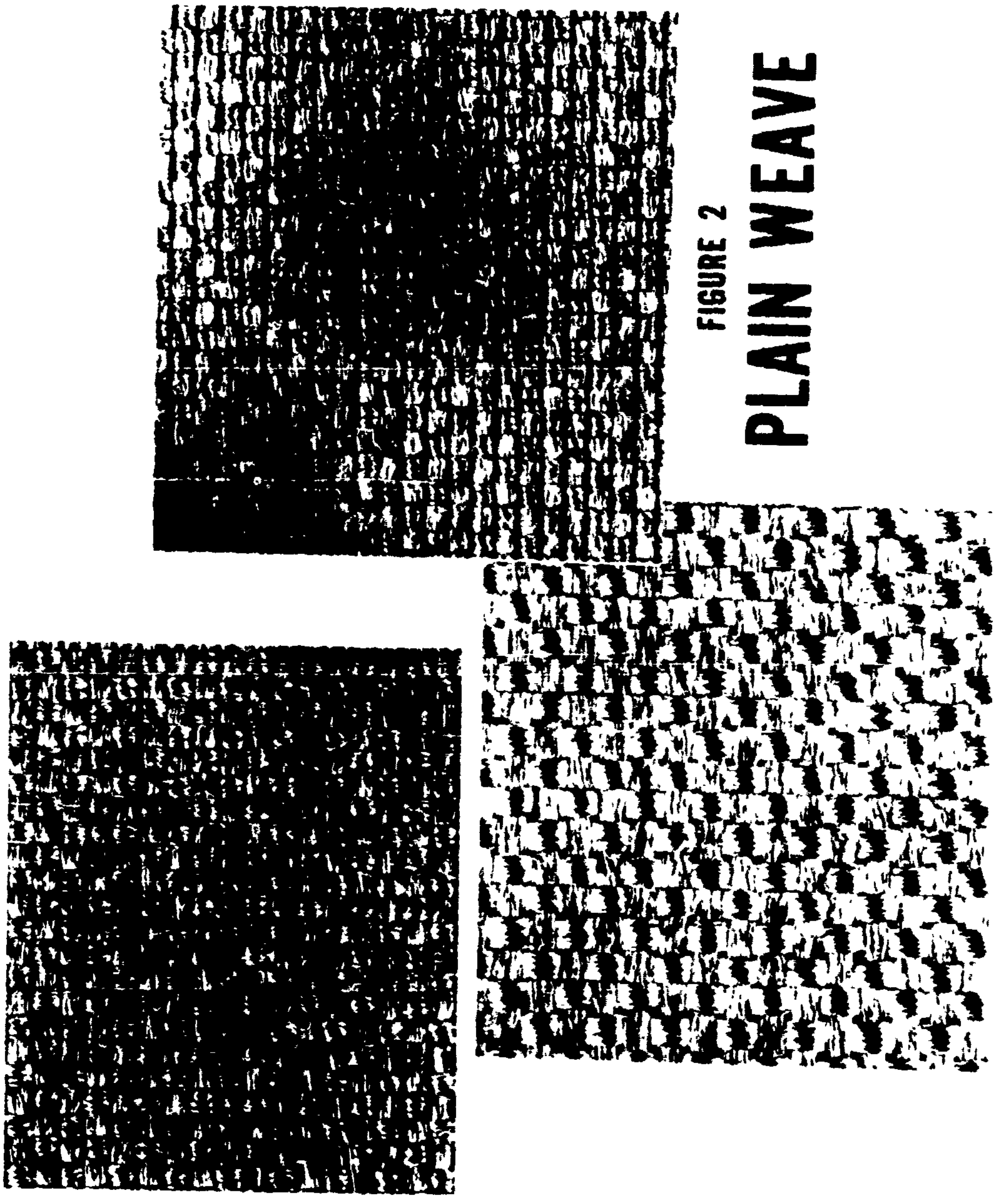

FIGURE 2
PLAIN WEAVE

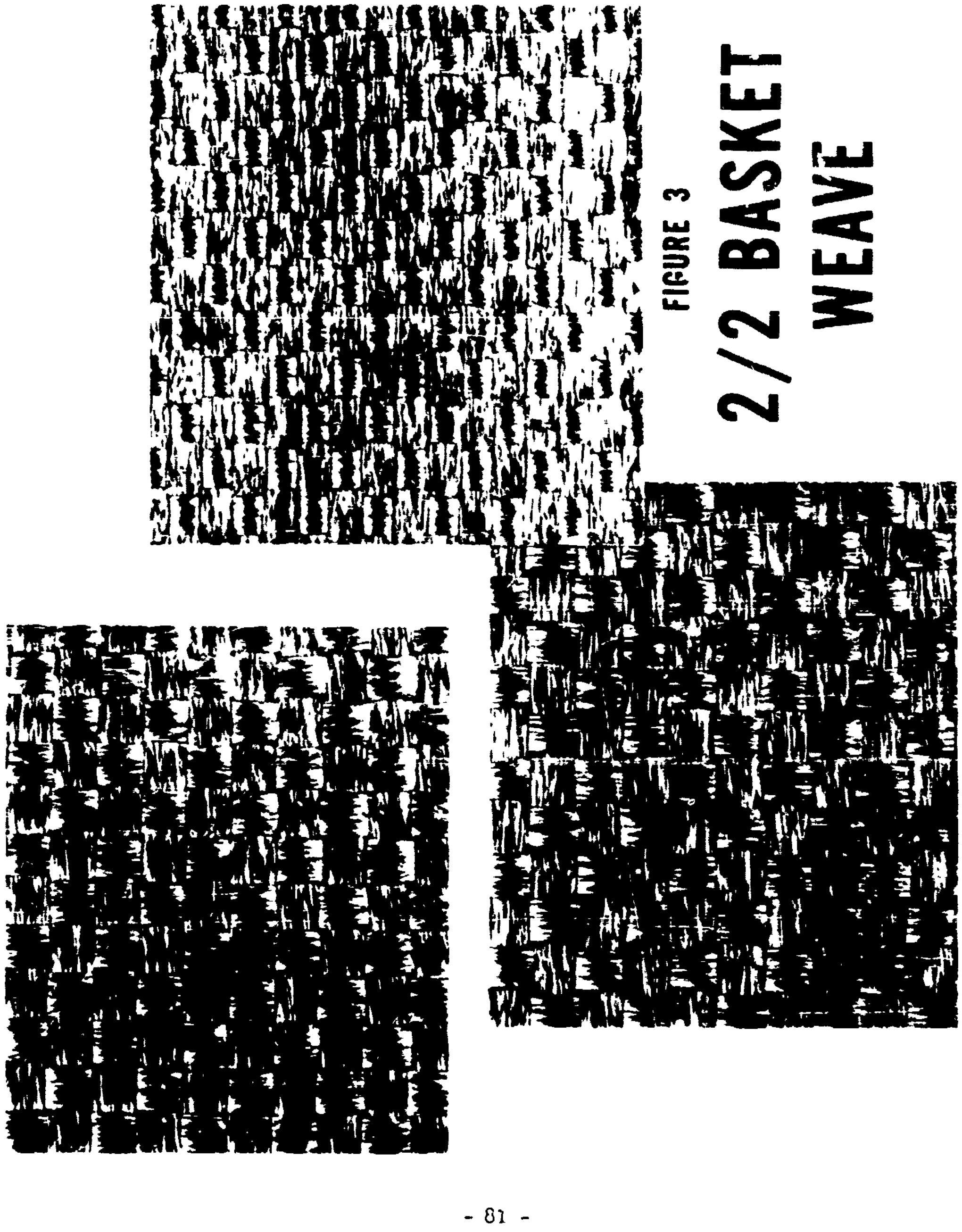

FIGURE 3
2/2 BASKET WEAVE

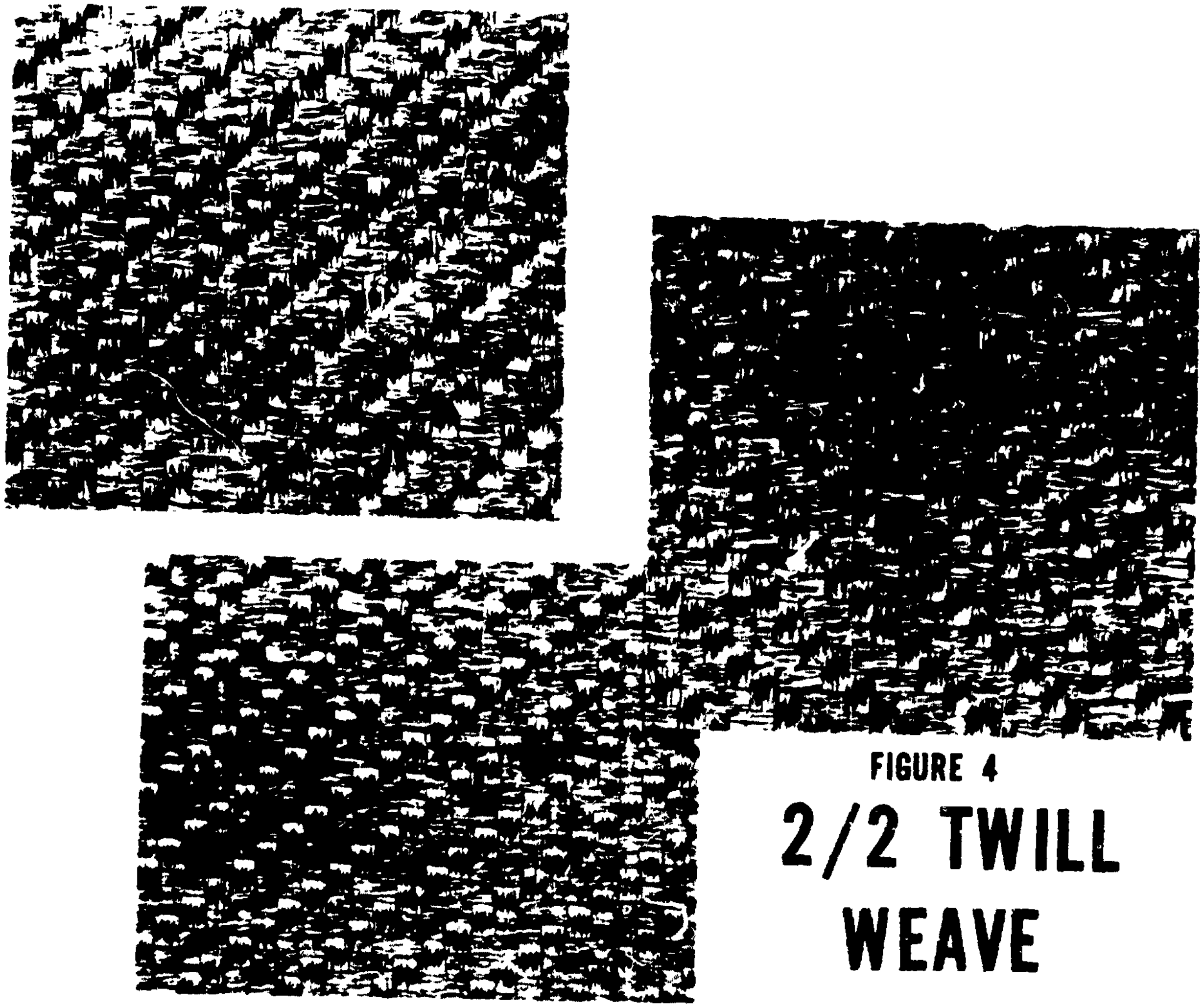

FIGURE 4
2/2 TWILL WEAVE

table 3 WRINKLE RECOVERY — FOLD ENDURANCE

Fiber Dia. Mils	Weave Design	Ends per Inch	Picks per Inch	Monsanto Wrinkle Recovery (Bent Against Curl) (Deg/180°) Warp	Monsanto Wrinkle Recovery (Bent Against Curl) (Deg/180°) Filling	Monsanto Wrinkle Recovery (Bent Against Curl) % Warp	Monsanto Wrinkle Recovery (Bent Against Curl) % Filling	MIT Folding Endurance (Cycles) Warp	MIT Folding Endurance (Cycles) Filling
0.5	Plain	61	61	56	41	31	23	723	758
	2 x 2 Basket	81	81	65	53	36	29	1037	962
	2/2 Twill	82	82	52	56	29	31	888	764
0.7	Plain	81	75	22	17	12	9	381	248
	2 x 2 Basket	82	119	20	28	11	15	409	466
	2/2 Twill	82	102	21	25	12	14	375	360
1.0	Plain	60	61	29	20	16	11	212	193
	2 x 2 Basket	81	81	38	22	21	12	230	234
	2/2 Twill	81	81	39	28	22	15	205	188
1.5 Mono	Plain	200	200	$\cong 0$	$\cong 0$	$\cong 0$	$\cong 0$	29.5	34.0
3.0 Mono	2/2 Twill	157	157	$\cong 0$	$\cong 0$	$\cong 0$	$\cong 0$	28.5	29.0

table 4 FABRIC MECHANICAL PROPERTIES

Fiber Dia. Mils	Weave Design	Ends per inch	Picks per inch	Yield Str. (lbs/in)		Rupture Str. (lbs/in)		Str. Translation Fib. to Fab. (%)		Tear Str. (lbs)	
				Warp	Filling	Warp	Filling	Warp	Filling	Warp	Filling
0.5	Plain	61	61	218		218	214	93.5	90.2	4.6	5.4
	2 x 2 Basket	81	81	213		226	206	95.8	87.0	▶12.5	▶13.0
	2/2 Twill	82	82	161		171	158	97.1	89.0	▶ 6.9	7.0
0.7	Plain	81	75	160		181	149	95.5	84.7	3.8	3.8
	2 x 2 Basket	82	119	160	226	184	242	95.9	87.5	7.5	▶10.0
	2/2 Twill	82	102	165	196	192	215	99.9	90.1	6.1	6.0
1.0	Plain	60	61			160	144	94.6	84.9	4.1	4.1
	2 x 2 Basket	81	81	208		217	193	95.6	85.1	8.3	8.3
	2/2 Twill	81	81	208		216	194	95.5	85.9	5.4	6.2
1.5 Mono	Plain	200	200			53	53			0.5	0.4
3.0 Mono	2/2 Twill	157	157			122	72			2.2	2.3

▶Tore Diagonally

table 4 FABRIC MECHANICAL PROPERTIES (cont.)

Fiber Dia. Mils	Weave Design	Ends per Inch	Picks per Inch	Yield Elongation (%)		Rupture Elongation (%)		Mod. of Elasticity (psi x 10^6)	
				Warp	Filling	Warp	Filling	Warp	Filling
0.5	Plain	61	61	4.6		7.6	3.0	6.5	9.0
	2 x 2 Basket	81	81	4.1		7.1	3.6	6.1	7.0
	2/2 Twill	82	82	5.7		9.1	3.3	5.3	8.5
0.7	Plain	81	75	11.9		16.8	4.5	1.5	4.9
	2 x 2 Basket	82	119	6.8	3.0	13.1	6.2	3.5	4.0
	2/2 Twill	82	102	6.3	3.1	13.8	6.9	3.2	5.0
1.0	Plain	60	61			8.3	3.0	3.0	7.1
	2 x 2 Basket	81	81	4.5		6.5	3.3	5.4	6.4
	2/2 Twill	81	81	5.2		6.6	2.7	4.5	6.5
1.5 Mono	Plain	200	200			13.3	7.9		
3.0 Mono	2/2 Twill	157	157			22.1	6.3		

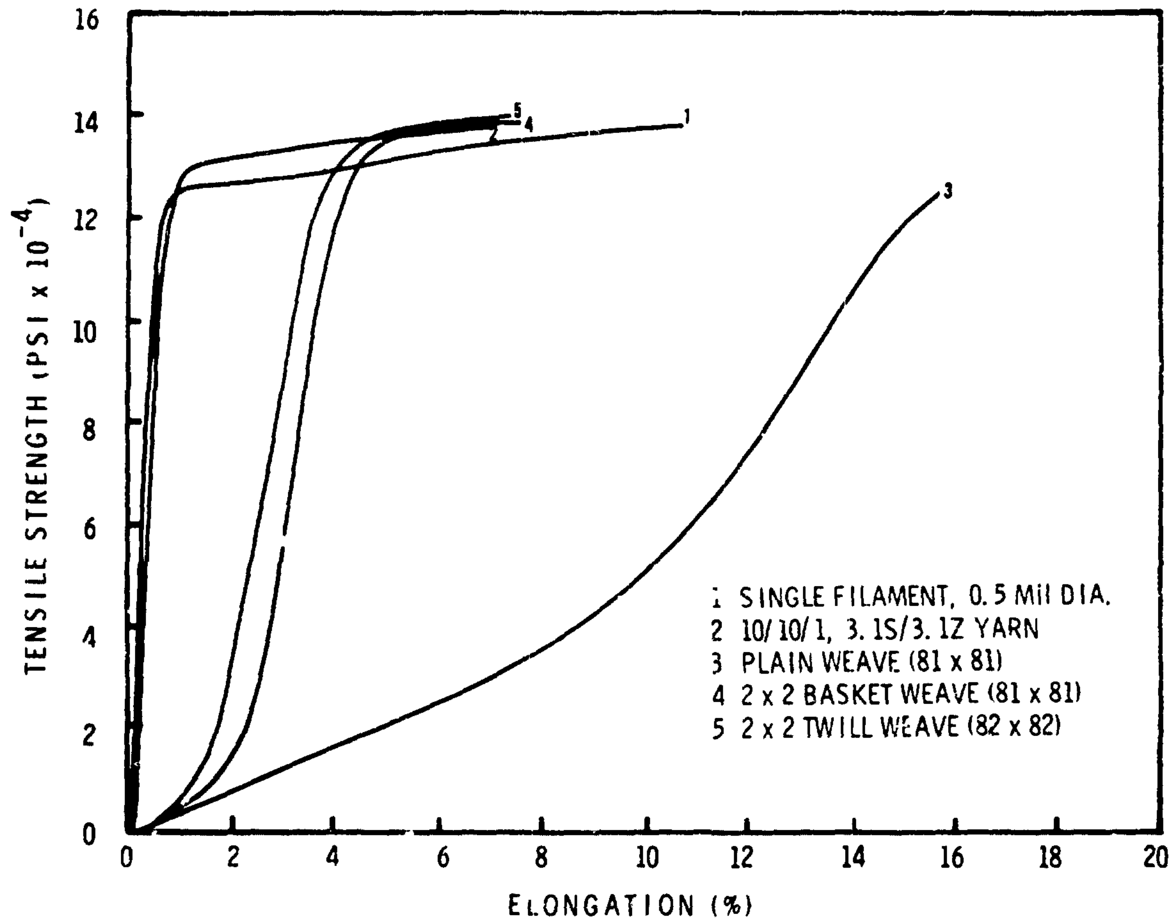

FIGURE 5 LOAD-ELONGATION CURVES FOR FIBER, YARN AND FABRIC

tension controls required for these ultra fine filaments.

II. COATINGS

The goal of this phase was to obtain a more temperature resistant, flexible, helium impermeable coating than is now available. Previous research (Ref. 2) did result in a coating which had good properties at temperatures to 1400°F, but as the temperature decreased in a given cycle, cracking occurred which caused an immediate increase in permeation of gas (in this case helium). This coating consisted of Dow-Corning S-2077 loaded with a ceramic frit having a low melting point. As the temperature passed the decomposition point of the S-2077, the frit fused and a thick, viscous coating formed which acted as a gas barrier. Upon cooling the coating should solidify and retain its impermeability, however flexing resulted in cracks as noted.

The specific requirements to be met in this research (Ref. 9), have been

a. To achieve thermal stability at 1500°F at a density equivalent to an altitude of 90,000 ft.

b. Achieve gas (helium) impermeability through a simulated re-entry heating cycle (max. flow of 2 cc/min/in^2).

The temperature range of 800° - 1100°F. became the primary problem area. At 800°F. (approx) the S-2077 started to decompose rapidly, however the frit does not fuse until a temperature of 1100° - 1200°F. is reached. This transition range represents a condition in which the gas permeability can most likely be very high.

Using the S-2077 as a base, various ceramic frits were added to achieve equal weight of elastomer and frit. After passing the mixture through a paint mill three or 4 times with the bite progressively closer, the percent solids was determined and diluted with toluene to 50 percent solids by weight.

The ceramic frit constitutes a major part of the final coating and the ability to maintain a flexible coating throughout a major portion of the heating cycle, especially at the 800°F. decomposition temperature of the S-2077, is dependent on the melting range of the frit. A total of eight frits, singly and in combination were evaluated using CS-105 type formulations. The particle size was controlled by milling in a three roll mill singly or in combination. The best single frit was that used in the coating labelled HTC-2 while the best two frit coating was that labelled HTC-52.

The experimental coatings were applied to a 200 mesh René 41 substrate woven of 1.6 mil wire, although stainless steel fabric of the same construction was used for some preliminary muffle furnace and permeability studies. A total of 84 variations of elastomer and frit were considered. In addition to the S-2077, a number of high phenyl polymers were compounded into S-2077 type formulations, but in all cases, degradation occurred, probably caused by the presence of a carbide formed as a result of carbon from the thermal breakdown of the phenyl groups. Brush coating was used and three applications were required to achieve a 7-10 ounce/sq.yd. coating. The coatings were cured for 15 min. at 480°F. after each application with a final cure at 480°F. for 16 hours.

The numerous coatings formulated were evaluated at temperature levels to 1500°F in order to establish helium leak rates. This initial screening made it possible to eliminate many coatings which at lower temperature levels did not leak, but as the temperature passed the critical 800° - 1100°F. range began to leak as the temperature was decreased to the point where the molten frit became solid and brittle, resulting in cracks where minute bends occurred. From this research two coating - frit combinations were found to be equal or superior to the original coating. (Ref. 2.). The leak rate over a given heating cycle is shown in Fig. 6, while Fig. 7 presents leak rates at 1200°F. for 15 minutes. The actual data used in Fig. 7 is shown in Table 6. As noted earlier this initial screening of coatings was conducted using metal fabric composed of mono-filament wire. Also as shown in the initial phase of this research, fabrics composed of multifilament yarns have far superior bending recovery, tear strength and fold endurance. Therefore, the final part of this research on coatings was to utilize fabrics with multifilament yarns as a base for the coating compounds. Since these multifilament fabrics were more tightly woven, it could be expected that some changes in the coating techniques would be necessary. From a study of application methods, such as dipping, rolling and brushing, and microscopic examination of coating penetration, it was determined that an initial brush coating using 25% solids in lieu of the 50% solids would result in good adhesion of the remaining coating layers. The CS-105, HTC-2 and HTC-52 were applied to a 2/2 twill composed of 49 filaments of 0.7 mil Chromel wire in an 80x80 construction. These three coated fabrics were then evaluated for helium leakage over the temperature cycle (Fig. 6) with results quite different, than obtained using a monofilament fabric. This could be attributed to tightness of weave, adhesion of coating, and multifilament yarn which promotes adhesion and gives good covering power.

In addition, the two best coating-frit combinations (HTC-2 and HTC-52) were evaluated for aerodynamic shear using the hot gas exhaust from an oxygen-hydrogen rocket motor. The high temperature, high velocity gas flow was passed parallel to the coated fabric to determine the effect on adhesion. The coatings were to be able to withstand a dynamic pressure of 14 pounds/ft.^{2} for 5 minutes at 1500°F. Actual test conditions were 67 lbs/ft^2 at 1500°F. Although the HTC-52 was better than the HTC-2 and CS-105, and least affected, few pinholes appeared in any of the specimens. The lighter weight coatings had a lesser tendency to crack and flake off than the heavier coatings and maintained a good degree of flexibility after cooling.

From this phase of research it was determined that a new coating-frit combination utilizing zirconium silicate and lead was found to be superior to the original coating frit (Ref. 1.). It has a low helium leak rate, flexibility, and is rubbery at room temperature. From Fig. 8, it can be seen that the use of multifilament yarn in the fabric substrate affords a marked reduction in leak rate over two other coating-frit combinations.

CONCLUSIONS:

As a result of this two pronged research program, it has been shown that superalloy fibers of ultra fine diameter can be twisted into a textile type yarn and woven into fabrics with most any weave pattern. Utilization of standard textile type equipment (slightly modified in the case of twisters) was proven feasible. Fabrics formed had superior mechanical characteristics as compared to monofilament fabrics. Two primary areas of future investigation include

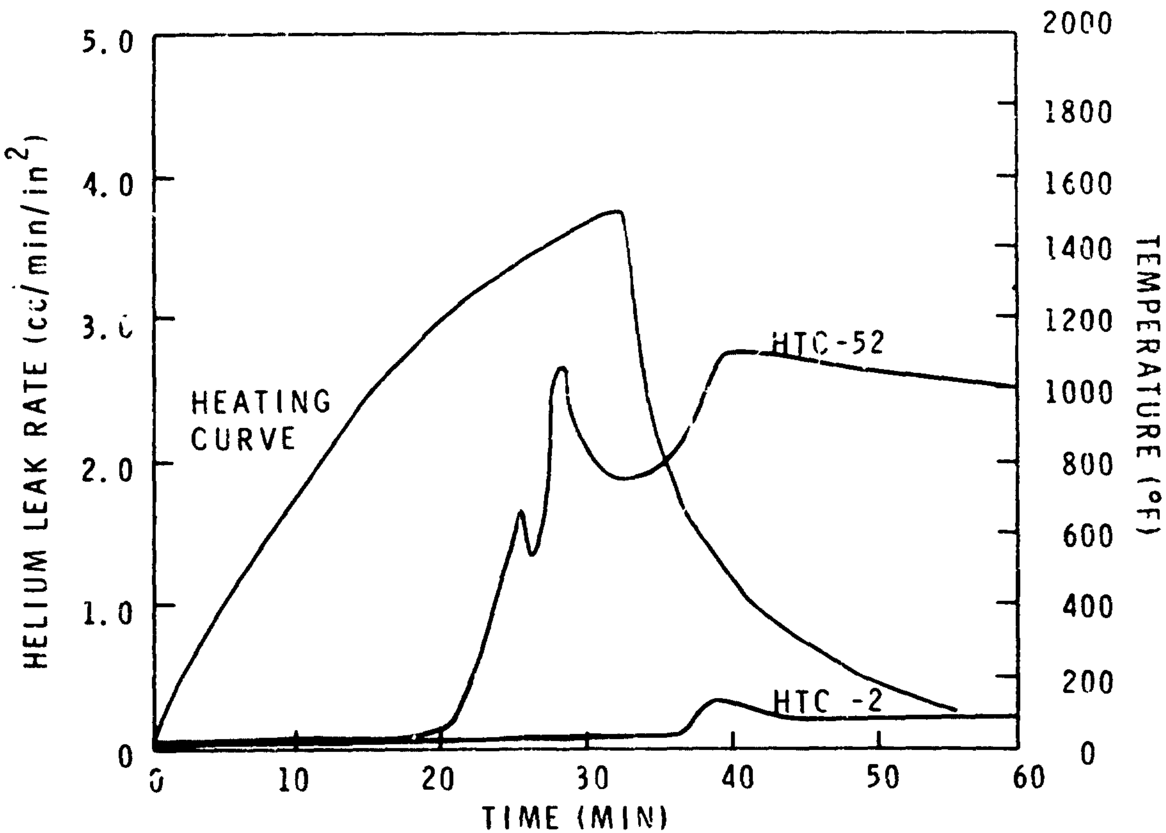

FIGURE 6 HELIUM LEAK RATE FOR TWO COATINGS

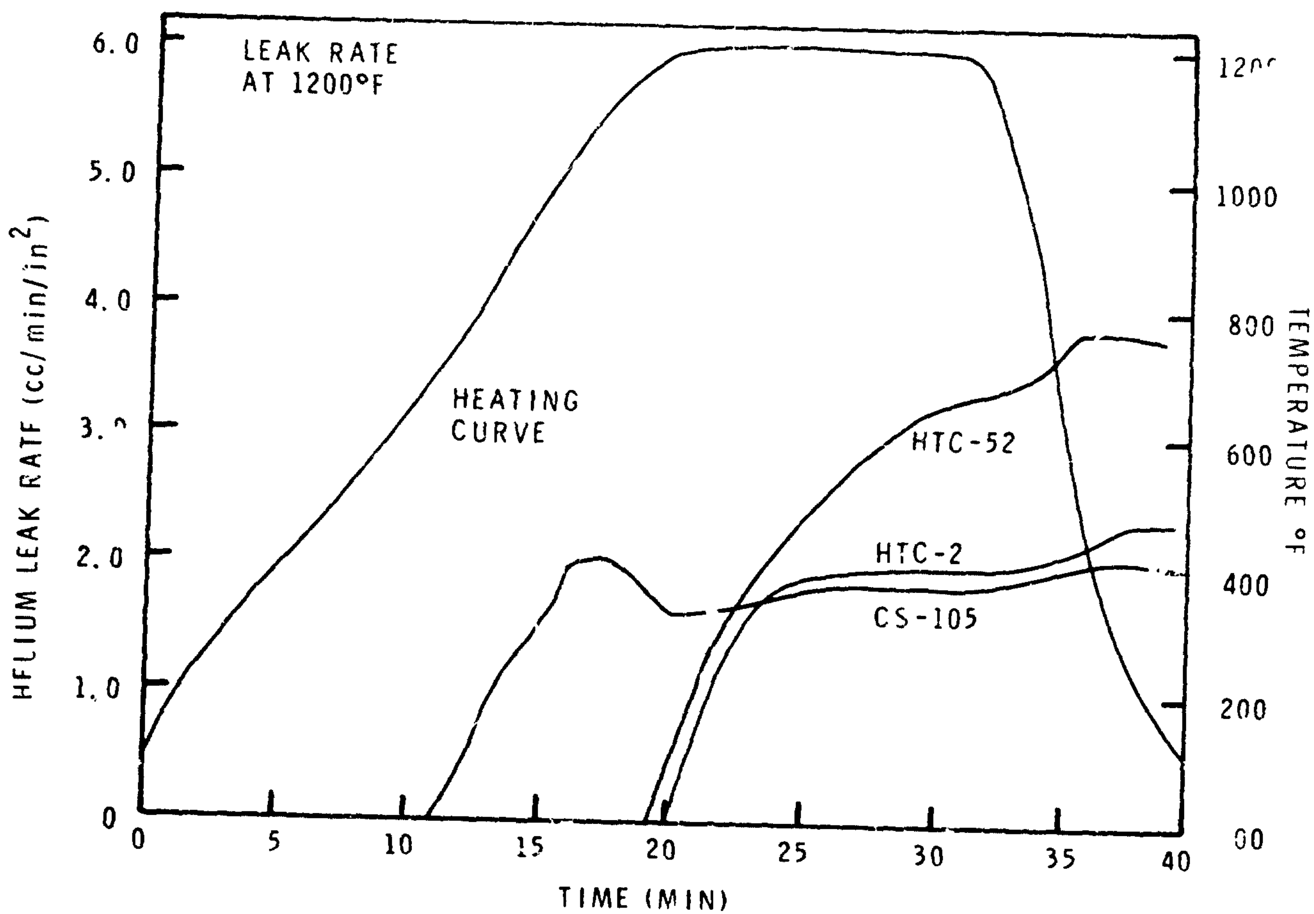

FIGURE 7 HELIUM RATE FOR MAXIMUM TEMP. (1200°F)

table 6 HELIUM LEAK RATE AT 1200°F

Coating Description	Weight (oz/yd^2)	Temp. (°F)	Time at Temp. (min.)	Leak Rate ($cc/min/in^2$)
CS-105	7.9	788		0.25
		1022		2.2
		1200		1.5
		1200	4	1.7
		1200	8	1.7
		1200	10	1.9
		662		2.0
HTC-2	6.9	1200		0.8
		1200	4	1.7
		1200	8	1.7
		1200	10	1.9
		644		2.1
HTC-52	8.7	1200		NO FLOW
		1200	4	2.3
		1200	8	3.2
		1200	10	3.3
		680		3.7

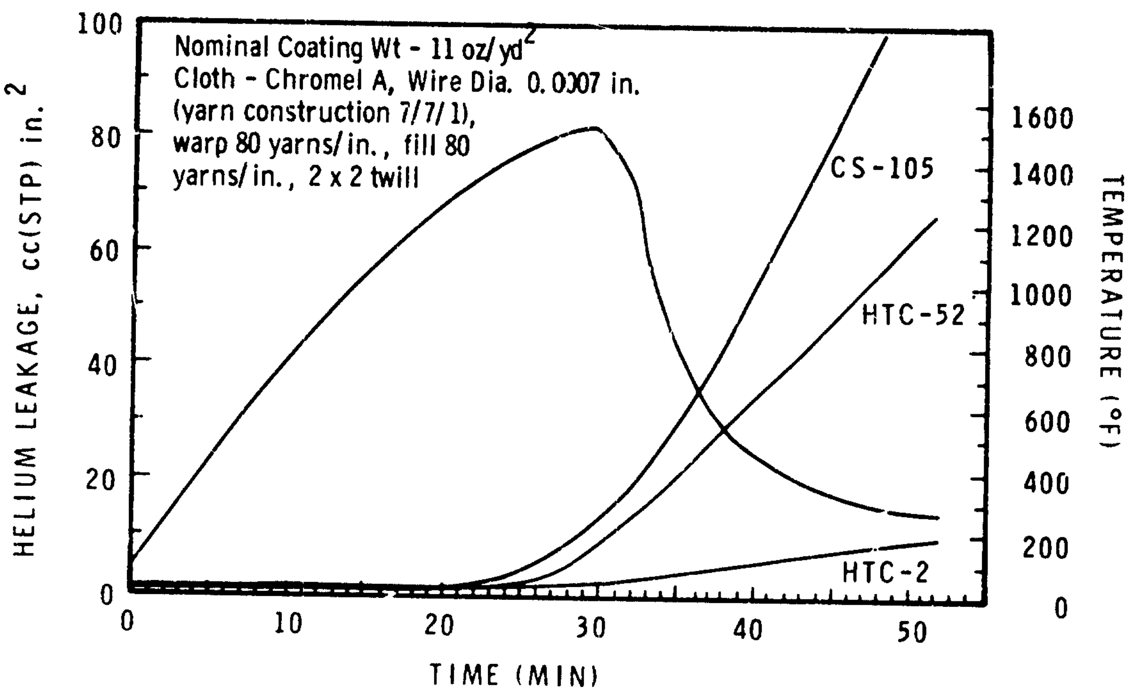

FIGURE 8 HELIUM LEAKAGE VERSUS TIME & TEMP. OBTAINED FOR SAMPLES HAVING NOMINAL COATING WEIGHT OF 11 OZ/YD^2 (Chromel A Cloth)

a. Geometry studies of double wall (airmat) fabrics of multifilament superalloy yarns.

b. High temperature characterization of multifilament metal fabrics.

In the case of coatings, improved coatings have been achieved which will withstand a typical simulated re-entry heating cycle with minimum helium permeability. Further research on coatings capable of enduring a re-entry cycle with a 2000°F. upper limit has been initiated and will utilize wholly new multifilament super-alloy fabrics with varying porosity to further investigate effect of porosity on helium permeability. The higher porosity fabrics are lower in weight and their successful use could result in lower overall weights in the final coating fabric composite.

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THEORETICAL AND EXPERIMENTAL INVESTIGATION OF METAL FABRIC EXPANDABLE STRUCTURES FOR AEROSPACE APPLICATIONS

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LIST OF SYMBOLS

A	in^2	cross section area of AIRMAT
E	lbs/in	Young's modulus of fabric
G	lbs/in	shear modulus of fabric
L	in	span of Delta AIRMAT
M	in lb	bending moment
R	in	radius of cylinder, also root chord length of a Delta wing
T	in lb	torsion moment
V	lbs/in	AIRMAT shear force per unit width
a, b	in	length and width of rectangular AIRMAT plate
h	in	thickness of AIRMAT plate
p	lbs/in^2	pressure
q	lbs/in^2	transverse loading of AIRMAT
s	in	circumference of AIRMAT cross section
w	in	deflection
x, y, z	in	coordinates
α, β	radians	angles of rotation of drop threads in xz- and yz- plane, respectively
γ	radians	shear angle
ϵ	in/in	normal strain
σ	lbs/in	membrane stress

SUBSCRIPTS

B	due to bending
S	due to shear
c	referring to core material of a sandwich plate
coll	collapse
wr	wrinkling

I. INTRODUCTION

Metal fabric expandable structures, including AIRMAT structures, appear to hold good promise for a number of aerospace applications. There are essentially three reasons for this: (1) the capability of fabric structures of being folded and packed into small spaces, (2) their good weight-to-strength ratio, and (3) in the case of reentry vehicles - their low wing loading and correspondingly low heat input.

Inherent in fabric structures are certain characteristics that cause their structural response to external loading in many cases to be different from conventional structures. The most important points are the following:

1. Woven fabrics in general are inhomogeneous aeolotropic materials, although many have, or may be considered as having orthotropic characteristics.

2. The stress-strain behavior of fabrics may or may not be elastic. This depends on many factors: on the basic material of the filaments and of the sealant, the build of the single thread, which may be monofilament or stranded; also on the weave pattern (plain, twill, satin, basket weave), the thread count, and the crimp in warp and fill direction. In the most general case, the stress-strain characteristics vary with magnitude, ratio, direction and previous history of the principal stresses.

3. A properly designed fabric structure is prestressed throughout by the pressure in biaxial tension and therefore able to take compressive stresses up to the level of the pressure stresses. When both stresses are equal incipient wrinkling will occur. Depending on the geometry and the loading condition of the fabric component, there is a greater or lesser margin between wrinkling and complete collapse. In a sense wrinkling and collapse of pressure-stabilized fabric structures correspond to crippling and elastic instability of conventional structures. The important difference is that fabric structures, upon wrinkling or even collapse, may recover as soon as the loading or part of it is removed, so that they remain intact for further operation.

4. A feature peculiar to pressurized fabric structures subjected to transverse loads is their ability to resist shear deformation by the mere presence of the gas pressure. An AIRMAT plate, for instance, behaves in this regard much as a sandwich plate does, the core of which has a shear modulus equal to the gage pressure of the AIRMAT.

5. Generally, the shear deflections of metal AIRMAT beams and plates under transverse loading are substantially greater than the bending deflections. In some cases the latter are almost negligible by comparison.

II. RESULTS OF A THEORETICAL AND EXPERIMENTAL INVESTIGATION OF METAL FABRIC EXPANDABLE STRUCTURES*

1. General

This study was conducted by Goodyear Aerospace Corporation under the sponsorship of the Aeronautical Systems Division. The study encompassed the following phases:

a. Experimental determination of the stress-strain characteristics of metal fabrics and AIRMAT in various biaxial stress conditions,

b. Methods of analysis of expandable metal fabric and AIRMAT structures, including literature survey,

c. Technology and fabrication (weaving of AIRMAT, spot welding techniques, coating procedure, etc.),

d. Tests of basic metal fabric and AIRMAT components for determination of their structural response to external loadings at various pressures, including correlation of test data and theoretical results.

2. Stress-Strain Characteristics of Metal Fabrics

Woven fabrics generally can be considered orthotropic materials. If they are perfectly elastic their stress-strain characteristics can be completely described by five elastic constants: the two Young's moduli along the axes of orthotropy, the corresponding Poisson's Ratios, and the modulus of rigidity. However, on elastic materials, Young's moduli and Poisson's ratios are known to be interrelated as follows:

$$E_x \cdot \mu_{yx} = E_y \cdot \mu_{xy} \qquad (1)$$

Thus, the number of elastic constants required actually is reduced to four.

The stress-strain characteristics were determined on a number of materials, mainly on Type 304 Stainless Steel fabrics and AIRMAT faces; for comparison, a one-ply Dacron-Neoprene fabric also was included.

The material specimens were partly of cylindrical shape; others were flat AIRMAT panels of 3 and 6 inch thickness. The specimens were inflated to various pressures, and at the same time longitudinal loads--either tension or compression--were applied; thus the strains in both principal directions could be measured over a wide range of biaxial stresses and stress ratios.

It was found most convenient to evaluate the results in the form of so-called "Everling" diagrams (Figures 1 through 5). In the Everling Diagram, the strains in the directions of the axes of orthotropy, which coincide with the principal stress axes (x and y) are plotted against each

* Reference 1.

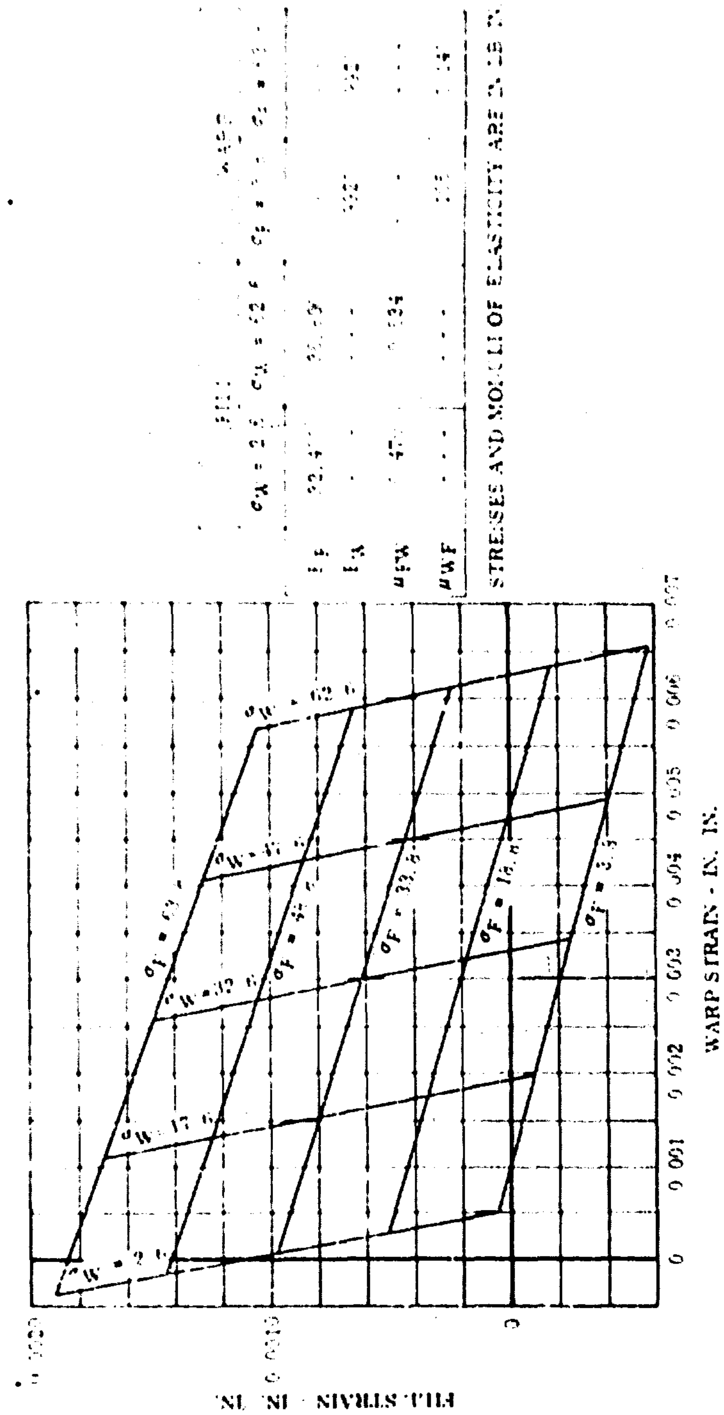


Figure 1. Everting Diagram for One-Ply SS304, 100x100x0.004 Plain-Weave Monofilament Fabric

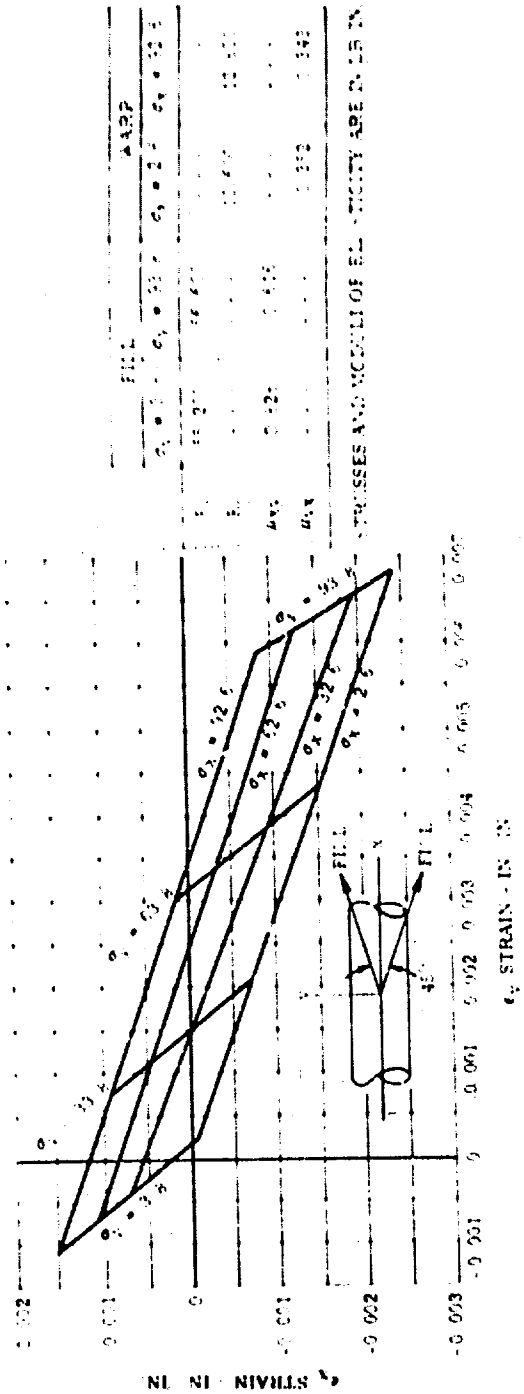


Figure 2. Everling Diagram for Two-Ply 33304, 100x100x0.0045 Plain-Weave Monofilament Fabric

	WARP		FILL	
	$\sigma_F = 2$	$\sigma_F = 32$	$\sigma_W = 2.4$	$\sigma_W = 32.4$
E_W	17,650	17,850	---	---
E_F	---	---	27,400	24,100
μ_{WF}	0.441	0.355	---	---
μ_{FW}	---	---	0.651	0.583

STRESSES AND MODULI OF ELASTICITY ARE IN LB/IN.

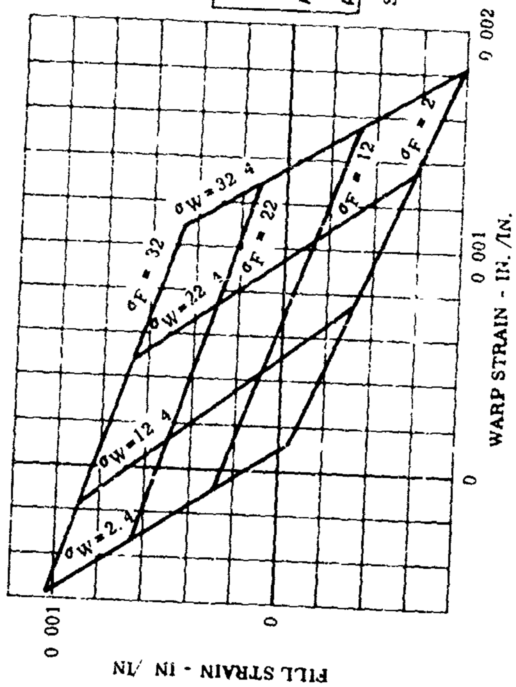
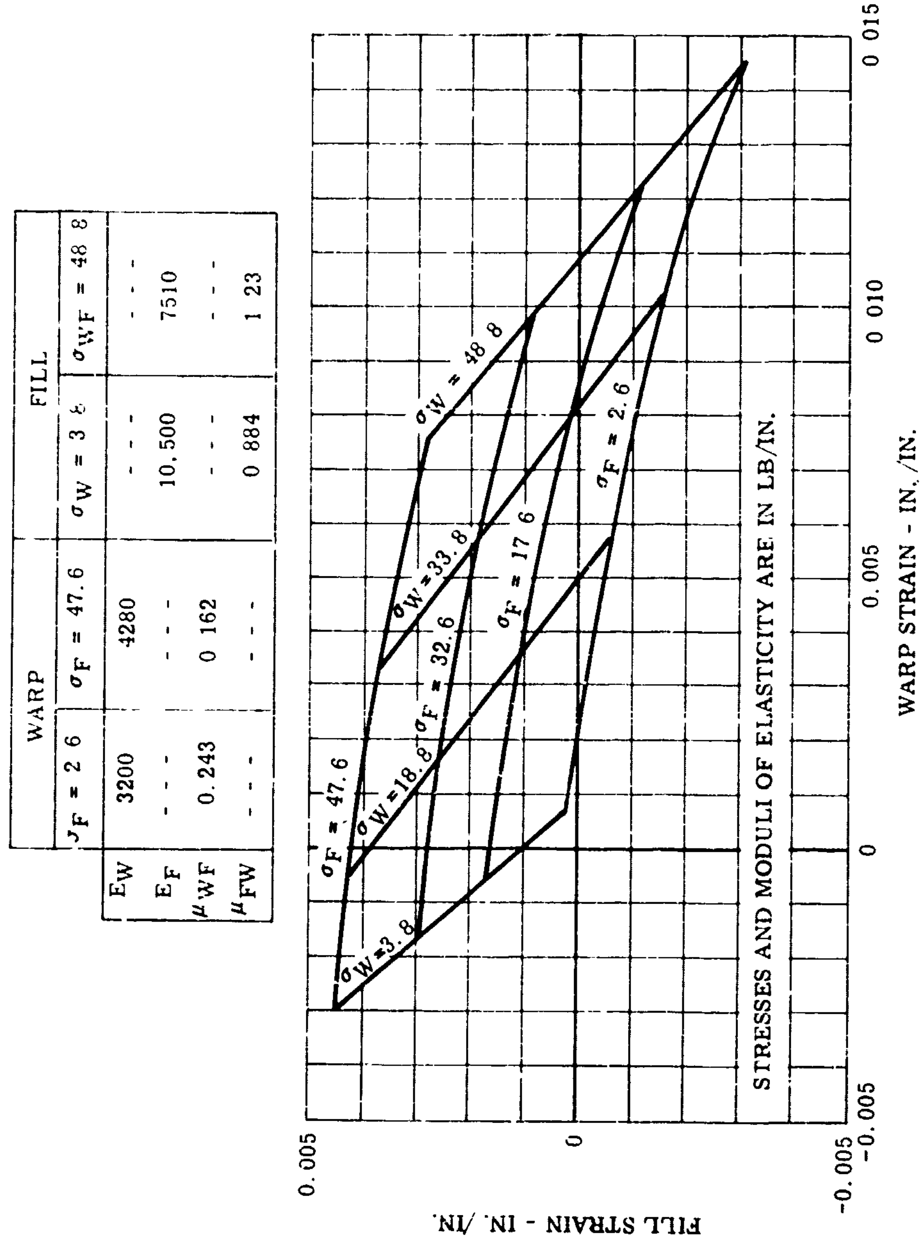


Figure 3. Everling Diagram for One-Ply SS304, 98x98x0.0045 ALMAT Skin (Monofilament, 3x3 Basket Weave)



	WARP		FILL	
	$\sigma_F = 2.6$	$\sigma_F = 47.6$	$\sigma_W = 3.8$	$\sigma_{WF} = 48.8$
E_W	3200	4280	- - -	- - -
E_F	- - -	- - -	10,500	7510
μ_{WF}	0.243	0.162	- - -	- - -
μ_{FW}	- - -	- - -	0.884	1.23

Figure 4. Everling Diagram for One-Ply Stranded-Yarn SS304, 100x100x7x0.0016 Plain Weave Fabric

	WARP		FILL	
	σ_F = 47.6	σ_F = 2.6	σ_W = 48.8	σ_W = 3.8
E_W	1710	1640	- - -	- - -
E_F	- - -	- - -	710	860
μ_{WF}	0.163	0.566	- - -	- - -
μ_{FW}	- - -	- - -	0.264	0.306

STRESSES AND MODULI OF ELASTICITY ARE IN LB/IN.

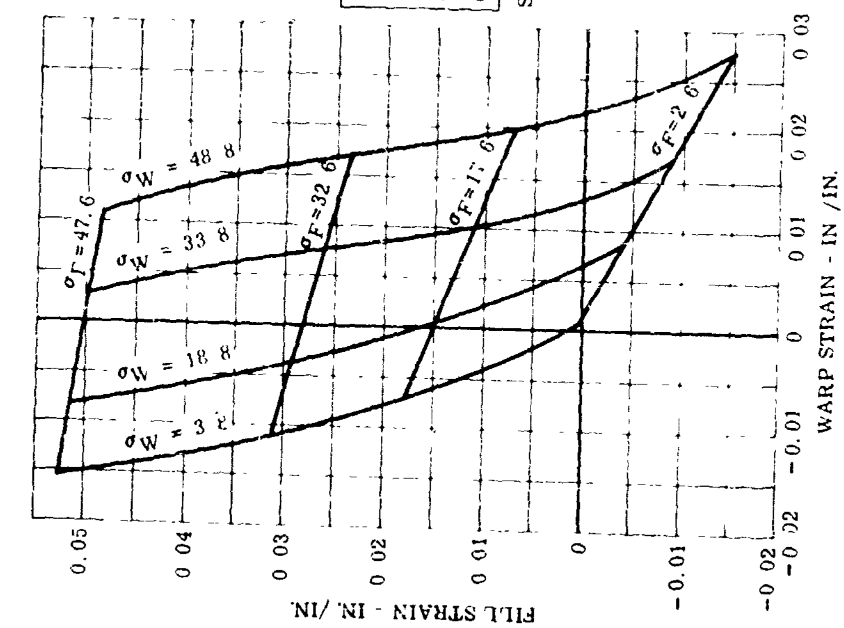


Figure 5, Everling Diagram for N363A10 Dacron Neoprene Fabric

other for various values of the stress parameters σ_x and σ_y. Thus two families of curves are obtained, and a simple calculation shows that the slope of the lines σ_x = constant, yields directly Poisson's Ratio μ_{yx} in the form

$$\partial \epsilon_x / \partial \epsilon_y = -\mu_{yx} \tag{2}$$

and similarly the slope of the lines σ_y = constant.

$$\partial \epsilon_y / \partial \epsilon_x = -\mu_{xy} \tag{3}$$

The E-moduli are found by dividing a stress increment into the corresponding strain increment:

$$E_x = \frac{\Delta \sigma_x}{\Delta \epsilon_{x\,(\sigma_y = \text{const})}} \tag{4}$$

and

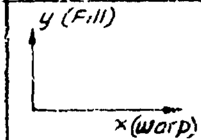
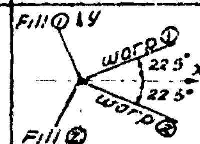
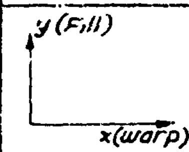
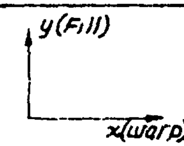
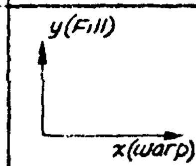
$$E_y = \frac{\Delta \sigma_y}{\Delta \epsilon_{y\,(\sigma_x = \text{const})}} \tag{5}$$

If each family of curves, σ_x = const. and σ_y = const, consists of parallel and equidistant straight lines the material is elastic within the test stress range.

For determination of the shear moduli, the cylindrical specimens were subjected to torsional moments, in addition to the longitudinal loads and the pressure stresses, and the twist angle over the length of the specimen was measured. Thus the G-modulus was obtained as function of all three stresses: longitudinal stress, hoop stress, and shear stress.

It should be noted here that all stresses are expressed in lbs per inch; correspondingly, the moduli have the same dimension. Some typical results for the six materials used are summarized in Table 1.

Table 1. Materials Description and Average Stress-Strain Characteristics

Material	Type of Fabric	Weight lbs/ft^2	Ply Orientation	E_x Average lbs/in	E_y Average lbs/in	G Average lbs/in
Type 304 Stainless Steel	Single-ply, plain-weave, 100x100x0.0045 monofilament	0.218		9,90[illegible]	34,600	114 to 286
	Two-ply, plain weave 100x100x0.0045 monofilament	0.425		13,200	55,900	6250 to 10400
Coating: S-2077	AIRMAT face, 3x3 basket weave 98x98x0.0045 monofilament	0.235		16,200	28,500	6 to 250
	Single-ply, plain-weave 100x100 stranded yarn 7x0.0016	0.222		3,750	9,000	5 to 450
Dacron-Neoprene	Single ply, N363A10	0.049		[illegible],680	790	30 to 100

3. Methods of Structural Analysis

A number of basic fabric components, such as cylinders, flat AIRMAT plates and beams, triangular plates with and without taper in thickness, and AIRMAT Delta wings have been analyzed for various loading conditions to obtain stress distribution and deflections and to define wrinkling and collapse criteria. The analysis was based on the material properties that had been determined on material specimens. The theoretical results were correlated to test data obtained on test cylinders and AIRMAT specimens--twenty in all--and, in general, good agreement was obtained.

As far as deflections of AIRMAT structures are concerned, the mechanism of shear transmission and the companion shear deflections require particular attention. Let us consider a cantilevered AIRMAT beam under uniform loading. The beam shall be wide enough that the effect of the rounded edges becomes negligible (Figure 6). It is assumed that small-deflection theory holds, so that bending and shear deflections can be linearly superimposed:

$$w \cong w_B + w_S \tag{6}$$

Then the loading of the beam can be resolved in a pure bending and a pure shear loading as indicated in the figure. In pure bending the beam deflects so that the drop threads remain normal to the AIRMAT face, i.e., each drop thread rotates by an angle α equal to the slope angle of the elastic line. In pure shear, the drop threads retain their original direction, but form an angle with the normal to the faces which equals the "shear angle" . As the originally rectangular element $h \cdot dx$ deforms into a parallelogram its volume per unit width decreases by

$$\begin{aligned} h \cdot f &= hdx(1 - \cos \gamma) \\ &\cong \frac{1}{2} hdx\, \gamma^2 \end{aligned} \tag{7}$$

where $\cos \gamma$ is replaced by the first two terms of the cosine series. Equating the energy required for this change in volume to the work performed by the shear force per unit width, V, gives

$$\frac{1}{2} phdx\, \gamma^2 = \frac{1}{2} V\, dw_S$$

or, with

$$\gamma = \frac{dw_S}{dx}$$

$$\frac{dw_S}{dx} = \frac{V}{ph} \tag{8}$$

For comparison, the shear deflection equation of a sandwich beam is

$$\frac{dw_S}{dx} = \frac{V}{G_c h} \tag{9}$$

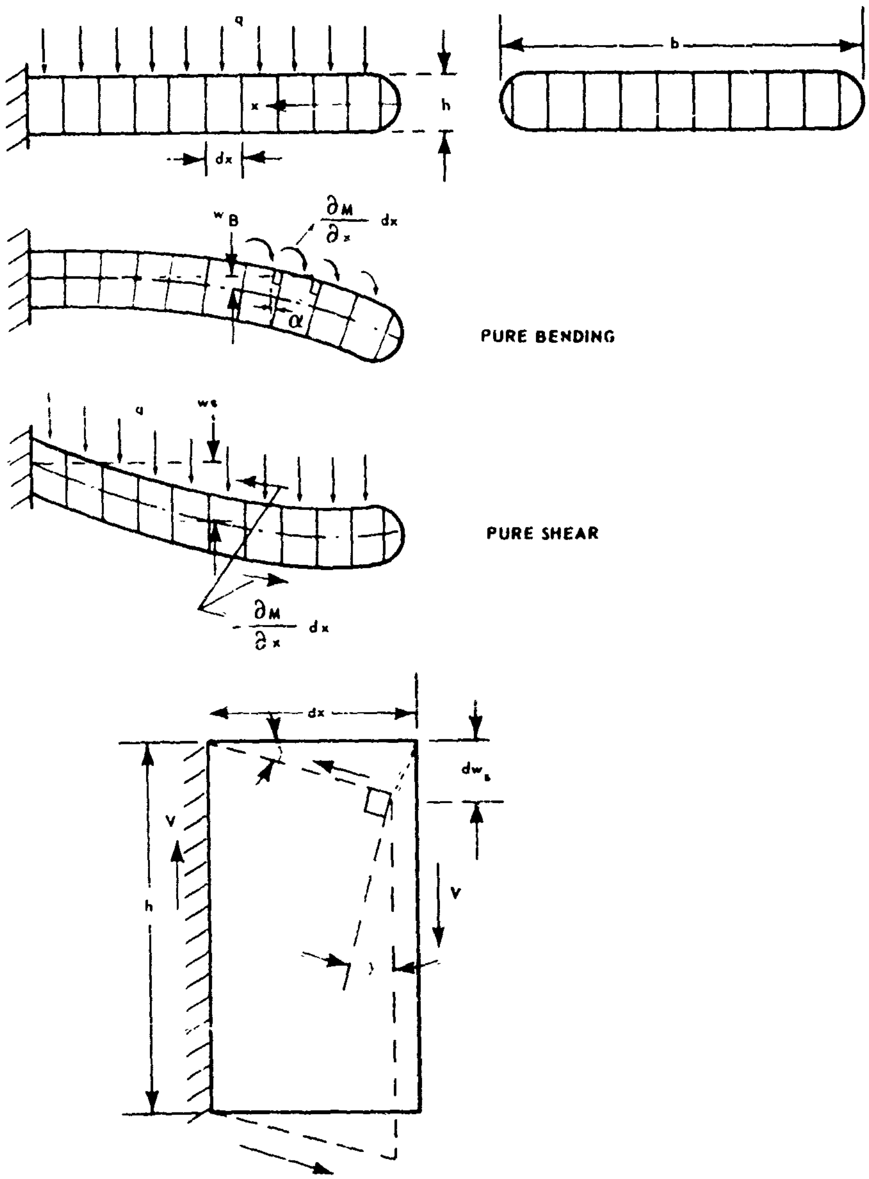


Figure 6. Bending and Shear Deflections of an AIRMAT Beam

where G_c is the shear modulus of the core material.

If an AIRMAT plate of arbitrary shape is subjected to non-uniform loading, with arbitrary boundary conditions, it is necessary to use orthotropic plate theory, with modifications to take the peculiarities of AIRMAT plates into account. McComb has solved this problem for rectantular plates (Ref. 2, 3), using the principle of minimum potential energy. In the case of AIRMAT plates, this energy consists of three parts: the strain energy in the AIRMAT faces, the energy associated with the volume change, and the potential energy of the external loading. McComb arrives at the following three differential equations:

$$\left.\begin{aligned} &ph\left(\frac{\partial \alpha}{\partial x} + \frac{\partial^2 w}{\partial x^2}\right) + ph\left(\frac{\partial \beta}{\partial y} + \frac{\partial^2 w}{\partial y^2}\right) + q_z = 0 \\ &\frac{h^2}{2}\left(A_{11}\frac{\partial^2 \alpha}{\partial x^2} + A_{12}\frac{\partial^2 \beta}{\partial x \partial y}\right) + \frac{h^2}{2}A_{33}\left(\frac{\partial^2 \alpha}{\partial y^2} + \frac{\partial^2 \beta}{\partial x \partial y}\right) - ph\left(\alpha + \frac{\partial w}{\partial x}\right) = 0 \\ &\frac{h^2}{2}\left(A_{21}\frac{\partial^2 \alpha}{\partial x \partial y} + A_{22}\frac{\partial^2 \beta}{\partial y^2}\right) + \frac{h^2}{2}A_{33}\left(\frac{\partial^2 \alpha}{\partial x \partial y} + \frac{\partial^2 \beta}{\partial x^2}\right) - ph\left(\beta + \frac{\partial w}{\partial x}\right) = 0 \end{aligned}\right\} \quad (10)$$

where the factors A are defined by

$$\left.\begin{aligned} A_{11} &= \frac{E_x}{1 - \mu_{xy}\mu_{yx}} \quad ; \quad & A_{22} &= \frac{E_y}{1 - \mu_{xy}\mu_{yx}} \\ A_{12} &= \frac{\mu_{yx} E_x}{1 - \mu_{xy}\mu_{yx}} \quad ; \quad & A_{33} &= G \end{aligned}\right\} \quad (11)$$

McComb calculated the mid-point deflection of a square AIRMAT plate, made of Nylon Neoprene and simply supported on all four sides, and found excellent agreement between theory and test data.

In the particular case of metal AIRMAT plates the bending deflections contribute even less to the total deflections than is the case with Nylon, due to the much higher modulus of elasticity of the stainless steel faces. This makes a simplification of McComb's equations possible, in that linear superposition of bending and shear deflections in general is valid without perceptible error. Thus

$$\left.\begin{aligned} w &= w_B + w_S \\ \text{and} \qquad \alpha &= -\frac{\partial w_B}{\partial x} \\ \beta &= -\frac{\partial w_B}{\partial y} \end{aligned}\right\} \quad (12)$$

and the original three variables (w, α, β) are reduced to two (w_B, w_S).

The differential equations for shear and bending deflections then can be written

$$\frac{\partial^2 w_S}{\partial x^2} + \frac{\partial^2 w_S}{\partial y^2} + \frac{q}{p\ h} = 0 \tag{13}$$

and

$$A_{11}\frac{\partial^4 w_B}{\partial x^4} + A_{22}\frac{\partial^4 w_B}{\partial y^4} + (2A_{12} + 4A_{33})\frac{\partial^4 w_B}{\partial x^2 \partial y^2} + \frac{2q}{h^2} = 0 \tag{14}$$

For a rectangular AIRMAT plate, simply supported along all four edges of length a and b, the solution is found by expanding both w_S and w_B into double Fourier series:

$$\left.\begin{aligned} w_S &= \sum_{m=1}^{\infty}\sum_{n=1}^{\infty} w_{mn_S} \sin\frac{m\pi x}{a}\sin\frac{n\pi y}{b} \\ w_B &= \sum_{m=1}^{\infty}\sum_{n=1}^{\infty} w_{mn_B} \sin\frac{m\pi x}{a}\sin\frac{n\pi y}{b} \end{aligned}\right\} \tag{15}$$

Substitution of Equations (15) into the differential equations (13) and (14) yields the unknown coefficients in explicit form:

$$w_{mn_S} = \frac{16}{\pi^4}\frac{q}{p\ h}\frac{1}{mn(\frac{m^2}{a^2} + \frac{n^2}{b^2})} \quad \text{(both m and n odd)} \tag{16}$$

and

$$w_{mn_B} = \frac{32}{\pi^6}\frac{q}{h^2}\left\{mn\left[A_{11}\frac{m^4}{a^4} + (2A_{12} + 4A_{33})(\frac{mn}{ab})^2 + A_{22}(\frac{n}{b})^4\right]\right\}^{-1} \tag{17}$$

In the case of an AIRMAT Delta wing, the Fourier series for w_S and w_B are written in a somewhat different form (Equations 19) to satisfy identically the boundary conditions, which read, for a uniformly distributed load q:

$$\left.\begin{array}{lll} \text{at } y = 0: & \dfrac{\partial w_B}{\partial y} \neq 0 \quad \text{and} & \dfrac{\partial w_S}{\partial y} = 0 \\[2ex] \text{at } y = L: & w_B = w_S = 0 & \\[2ex] & \dfrac{\partial w_B}{\partial y} = 0 \quad \text{and} & \dfrac{\partial w_S}{\partial y} \neq 0 \end{array}\right\} \tag{18}$$

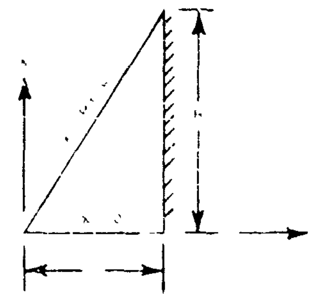


Figure 7. AIRMAT Delta Wing

$$\left.\begin{array}{l} w_S = \displaystyle\sum_{m=0}^{\infty} \sum_{n=0}^{\infty} A_{Smn} \cos\left(\frac{m\pi x}{R}\right) \sin\left[\frac{(4n+1)\,\pi\,(L-y)}{2L}\right] \\[3ex] w_B = \displaystyle\sum_{m=0}^{\infty} \sum_{n=0}^{\infty} A_{Bmn} \cos\left(\frac{m\pi x}{R}\right) \left[1 - \cos\left\{\frac{(4n+1)\,\pi\,(L-y)}{2L}\right\}\right] \end{array}\right\} \tag{19}$$

Variation of the potential energy and integration in this case leads to two systems of simultaneous linear equations in the unknowns A_S and A_B.

Figure 8 illustrates the calculated tip deflections of a stainless steel AIRMAT Delta wing under uniform loading and at three different gage pressures in comparison to the test data.

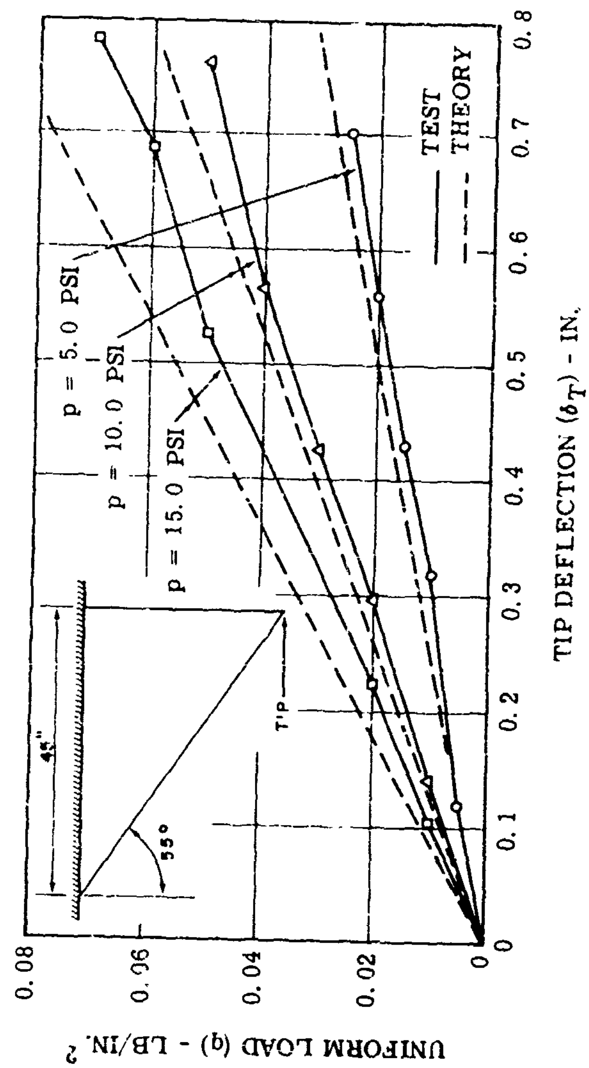


Figure 8. Tip Deflection of a Delta Wing under Uniform Loading

Wrinkling and Collapse Criteria

Table 2 gives the theoretical wrinkling and collapse criteria for fabric cylinders and AIRMAT beams or plates when subjected to bending, or torsion, or a combination of both. These criteria may be considered lower bounds, primarily because the fabric or AIRMAT face is considered a pure membrane, with no inherent flexural stiffness. Collapse tests on cylindrical as well as on AIRMAT specimens revealed that the first test consistently gave higher collapse moments than the theory predicted; in subsequent collapse tests on the same specimen, the actual collapse moment approached the theoretical value fairly close, apparently because the creases produced in the first collapse test had eliminated much of the flexural stiffness of the fabric.

Figure 9 exemplifies this phenomenon on a pressurized stainless-steel fabric cylinder subjected to pure bending moments. The collapse parameter at the first test, conducted at 2 psi gage pressure on the virgin cylinder was found to be

$$M_{coll}/p\pi R^3 = 1.40$$

while the theoretical value is 1.0. The subsequent tests at 4, 6, 8, and again at 2 psi pressure yielded collapse parameters between approximately 1.0 and 1.2.

4. Technology of Metal Fabric and AIRMAT Structures

a) AIRMAT Weaving

The AIRMAT required for fabrication of the test specimens was woven on GAC's pilot loom. AIRMAT of three and six inches thickness was produced; a third type tapered in thickness from 6 to 3 inches over a 48 inch length.

In an experimental weaving phase, several weave patterns (plain weave, twill, satin, and basket weave) were tried. A 3x3 basket weave was found to be the most satisfactory. The thread count is 98 in both warp and fill directions. The number of drop threads is 31 per square inch. Warp, fill, and drop wires have the same 0.0045 inch diameter. Since the crimp is substantially reduced in comparison with a plain weave of equal thread count, the cloth appears somewhat coarser; it is also more sleazy, as there is less inter-locking of the wires.

The drop threads are actually additional warp wires that cross from one face to the other, at a spacing of less than 0.2 inch.

b) Spot-Welding of Stainless Steel Cloth and AIRMAT Faces.

Highly efficient seams are of primary importance for the over-all efficiency of metal fabric and AIRMAT structures.

After some experimenting, a two-row seam with 1/20 inch spot spacing and 3/32 row spacing was adopted. The seams were produced by an automated

Table 2. Wrinkling and Collapse Criteria

	Load Condition	Wrinkling Criterion		Collapse Criterion	
CYLINDERS	Pure Bending	$M_{wr} = \frac{1}{2}\pi p R^3$	(20)	$M_{coll} = \pi p R^3$	(21)
	Pure Torsion	$T_{wr} = \sqrt{2}\,\pi p R^3$	(22)	$T_{coll} = \sqrt{2}\,\pi p R^3$	(23)
	Combined Bending and Torsion	$\frac{M}{M_{wr}} + \left(\frac{T}{T_{wr}}\right)^2 = 1$	(24)	$\frac{M}{M_{coll}} + \left(\frac{T}{T_{coll}}\right)^2 = 1$	(25)
FLAT AIRMAT	Pure Bending	$M_{wr} = p A^2/s$	*(26)	$M_{coll} = p A R$	*(27)
	Pure Torsion	$T_{wr} = 2 p A\sqrt{AR/s}$	*(28)	$T_{coll} = 2p A\sqrt{AR/s}$	*(29)
	Combined Bending and Torsion	$\frac{M}{M_{wr}} + \left(\frac{T}{T_{wr}}\right)^2 = 1$	(30)	$\frac{M}{M_{coll}} + \left(\frac{T}{T_{coll}}\right)^2 = 1$	(31)

*
A = cross section area
2R = thickness
s = circumference of cross section

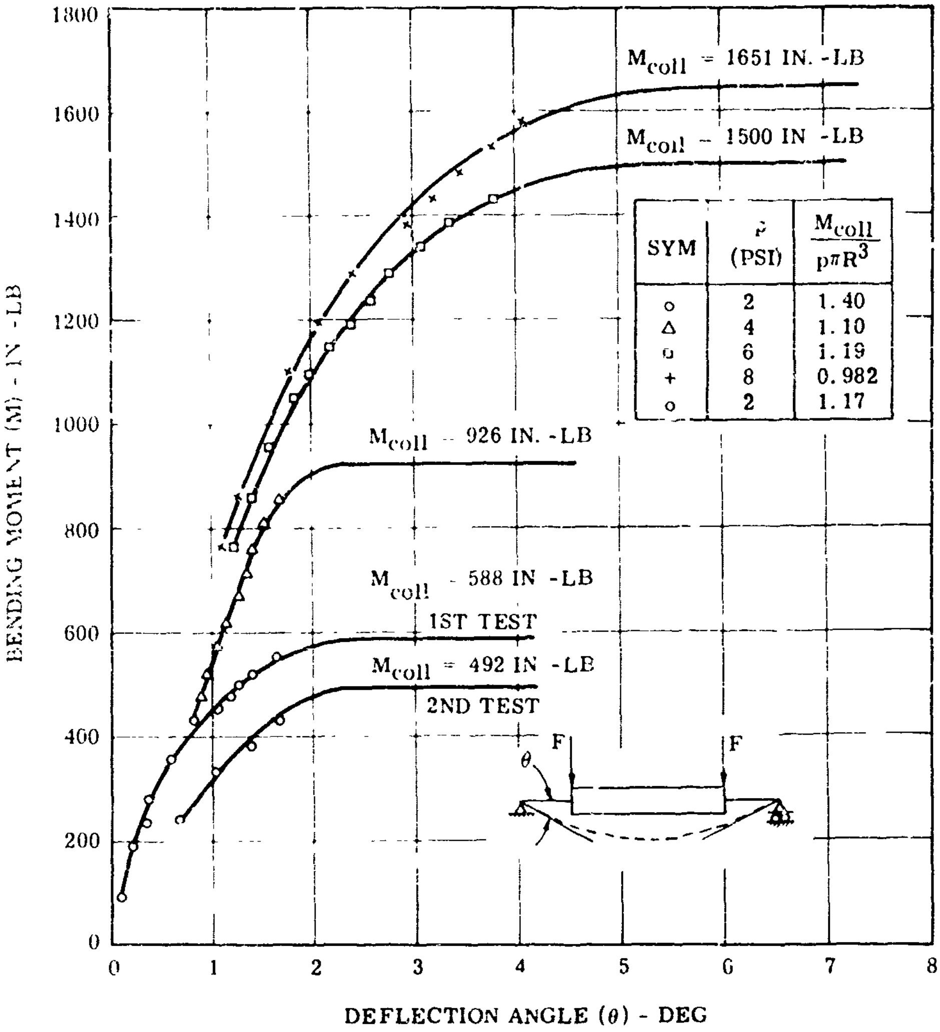


Figure 9. Collapse Bending Moments of a Two-Ply Stainless-Steel Fabric Cylinder (8 in. Diameter)

roll spot welder with pressure control, automatic drive mechanism and firing timer.

The efficiency of this seam is particularly good when used across the fill direction of the standard plain-weave cloth. Here the seam strength is approximately 97 percent of the strength of the parent material. Across the warp direction of the cloth, and also on AIRMAT faces in either direction, the seam efficiency drops off to between 60 and 70 percent. Apparently the high warp crimp in the plain-weave cloth and the less homogeneous texture of the basket-weave AIRMAT faces are responsible for these lower figures.

On AIRMAT components that require an extra cover ply it is necessary to ply-weld the cover ply to the AIRMAT face in a fairly narrow pattern, in order that both plies be in good contact and allow the sealant, which is applied from the outside, to strike through both plies and to form a positive gas barrier on the inside. Ply-welding was performed on the same automated roll spot welder. The weld spots were arranged in a pattern of equilateral triangles with approximately 0.6 inch side length.

c) Coating of Metal Fabric and AIRMAT Components.

The Silicon-rubber elastomer S-2077 was used as a sealant on all stainless-steel specimens. Application by brush was found to be preferable to spray-coating. Usually five brush coats were applied; however, very long cylinders, which made brush coating on the inside impractical. were successfully sealed by three spray coats on the inside and three brush coats on the outside. Between coats, the specimens were allowed to air-dry, then precured for 15 minutes at 500°F. The final cure required 16 hours at 500°F.

The upper limit of the heat resistance of the S-2077 elastomer was found to be 800°F. At this temperature, crazing of the sealant may occur after one or two temperature cycles and cause leakage of the fabric or AIRMAT component. A number of specimens, cylinders as well as AIRMAT components, were tested at elevated temperature. The temperature was kept at a 725°F maximum in order to preclude premature leakiness. The effect of the elevated temperature upon the deflection characteristics was moderate.

III. CONCLUSIONS AND RECOMMENDATIONS

The structural response to short-time loading of metal fabric structures can be predicted with sufficient accuracy. Appropriate analytical methods for determination of deflections, stress distribution, wrinkling and collapse conditions, and structural failure are available.

More information should be gained on load-time effects, such as creep, hysteresis, and previous stress history, particularly at elevated temperature.

For certain applications, fabrics and AIRMAT woven from multi-filament metal yarns instead of wires appear to offer advantages, especially with regard to tear resistance and foldability. Further research on this type of fabric is considered of great importance.

For AIRMAT structures that require the best stiffness characteristics possible, it has been suggested to use AIRMAT with canted drop threads. Preliminary theoretical investigations and successful experimental weaving, on the GAC loom, of this type of AIRMAT are encouraging.

The efficiency of the spot-welded seams was satisfactory for the purpose of the subject study. Further improvement for actual applications is desirable and may be possible.

In summary it can be said with confidence, that metal fabric and AIRMAT structures can be developed to the high degree of reliability and efficiency that is required for aerospace applications.

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LIMITED WAP APPLICATIONS
FOR EXPANDABLE STRUCTURES

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LIMITED WAR APPLICATIONS
FOR EXPANDABLE STRUCTURES

The President of the United States has reiterated on many occasions the fact that the United States will give its utmost support to the free nations of the world in their battles against Communism. The President, in effect, was saying that the United States would not stand aside and see freedom die in small countries around the world without a fight. At the time he pledged that support, he listed the three types of conflict that could be envisioned.

At the top of the list, of course, is the nuclear holocaust; next are limited wars; and finally insurgent actions within a specific country. The support he was offering followed earlier statements by Premier Khrushchev, who had listed the same three types of conflict and had threatened that if in the propagation of Communism it was necessary to engage in any of those conflicts, Russia would deal itself in.

First let us consider some definitions. Limited war is one that is limited in some aspect, such as geographical area, types of troops, or the method of fighting. Examples of this type of war are the Korean action, the Indo-China war, and the most recent India-Chinese skirmishes. Each of these were confined to comparatively small areas and at no time threatened to extend into a world conflict. Of more limited character is the insurgent action, whereby some elements within a country attempt to overthrow the existing government by military force.

Since that time, limited wars and insurgent actions have broken out in increasing numbers, and countries around the world have sought, and been given, U. S. support to preserve their way of life.

In furnishing this support we have found that instead of an advanced type of conflict, the free world is engaged in numerous small-scale, old-fashioned, brush-fire wars. We found that these small wars could be prosecuted to better advantage with WW II weapons and equipment, than with the latest technological systems that had been designed primarily for the wholly unwanted all-out nuclear war. Most of the conflicts have been in jungles or other remote areas where it was almost impossible to extend supply lines or provide air cover. We soon realized that a change had to be made in weapons and tactics if the support we were giving was to be of value.

The United States reacted by setting up the AF Systems Command's Limited War Management Office, special air warfare centers, and various swift-strike groups within the armed services.

My discussion is concerned with just one area in the study of new concepts and techniques to prosecute both limited wars and special air warfare actions. And for this discussion, we can consider limited war and special air warfare as the same thing.

I am here to discuss site support, including such items as shelters and logistics, and in particular, expandable structures.

LIMITED WAR SUPPORT REQUIREMENTS:

Any item or concept being considered for application to limited war support should meet certain limited war requirements. For evaluation of the applicability of expandable structures to limited war, we must first consider the support requirements. See Figure 1.

<u>Environment</u>

A limited war may be started in any part of the world. The design criteria for support items must include environmental considerations that are global in nature. However, there are certain areas of the world, by the nature of the terrain and environmental conditions, that are more likely to support an insurgency or limited war action. Areas that provide good cover or camouflage such as jungles, delta areas, or inaccessible portions of a country are more likely to support these actions than desert or arctic conditions. Therefore, for the most part, our design can be based on temperature conditions, rather than on extremes of environment.

<u>Critical Support Areas</u>

The critical support areas for limited war or counterinsurgency actions usually take place in the most remote areas of a country. This is the place where hostile forces or insurgents usually gather to avoid detection. From this area, the hostile action usually begins. In larger or very inaccessible countries, the amassing of insurgents may happen at several dispersed points.

The conflict would therefore take place at these remote points when the enemy forces first meet resistance. Government troops are usually airlifted by means of helicopters to these areas to combat insurgents or hostile groups. As the battle is taking place in these remote sites, troops, supplies, arms, and other materials become critical support items.

Other critical support areas are those requiring emergency aid. This could be natural disasters, such as earthquakes, emergency civil disorder, or to aid stranded personnel whose aircraft have been downed in remote areas.

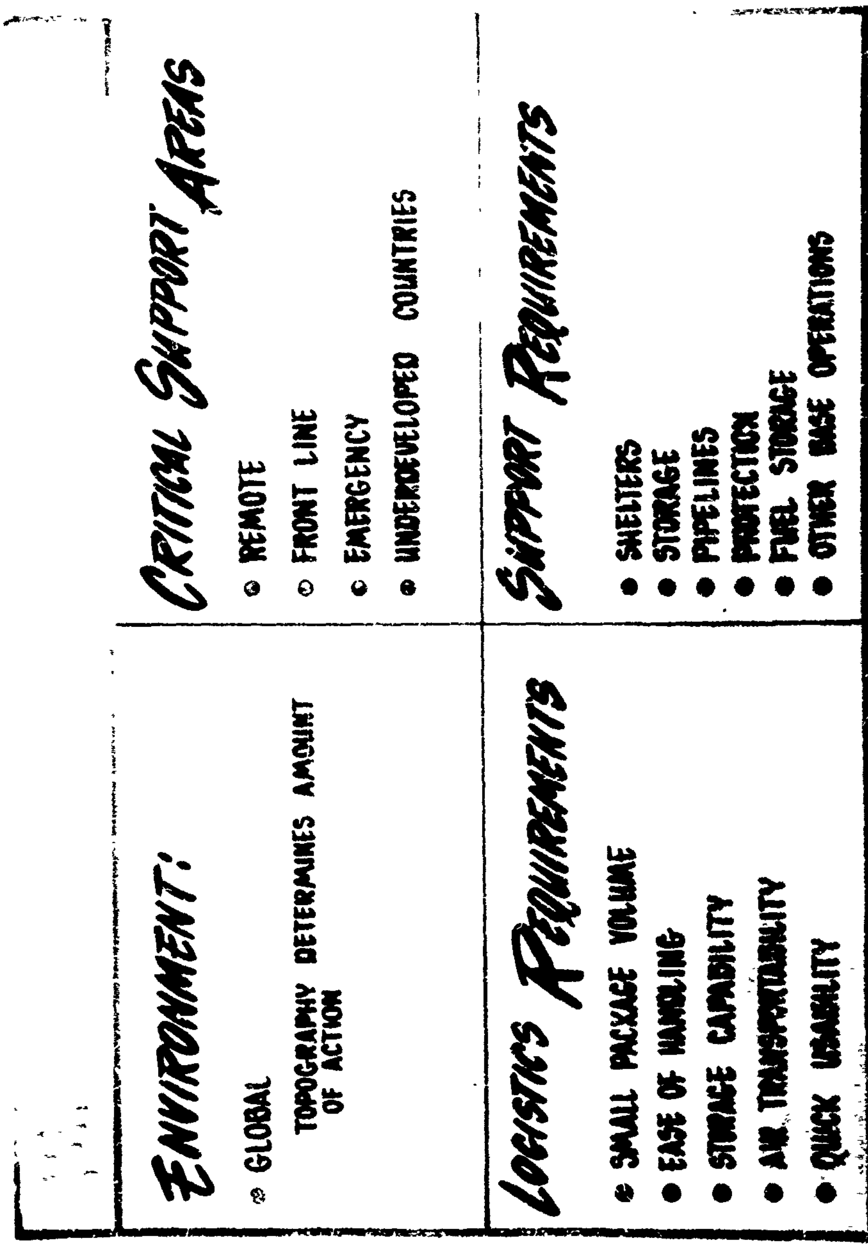
ENVIRONMENT:
GLOBAL
TOPOGRAPHY DETERMINES AMOUNT OF ACTION
CRITICAL SUPPORT AREAS
REMOTE
FRONT LINE
EMERGENCY
UNDERDEVELOPED COUNTRIES
LOGISTICS REQUIREMENTS
SMALL PACKAGE VOLUME
EASE OF HANDLING
STORAGE CAPABILITY
AIR TRANSPORTABILITY
QUICK USABILITY
SUPPORT REQUIREMENTS
SHELTERS
STORAGE
PIPELINES
PROTECTION
FUEL STORAGE
OTHER BASE OPERATIONS

Figure 1

The remote, front line areas which need the largest volume and weight of supplies remain the most critical support areas. Those peaceful countries which are underdeveloped and have no adequate transportation system can also be listed as an area for major supply concern.

Logistic Requirements

The conflict occurs, as stated before, in remote areas that support insurgent gatherings. As such, these are usually away from main supply lines, and not connected to good transportation facilities.

To adequately support any counteraction or drive, the proper material must be supplied. To enhance cargo efficiency, the supplies should be packaged in a small volume, and should not be unwieldy. This includes the formation of proper packaging and shaping concepts. The material must also be capable of being stored for long periods of time.

It must be air transportable to the inaccessible areas, and the material or system must be able to be quickly utilized on arrival. System reusability is dependent on new costs of a system versus the labor and effort to repackage the original item. Some systems need destruction capability in forward "no-man's land" where control of these areas are in doubt from day to day. All these logistics requirements are desirable in any situation, but in limited war, and counterinsurgency actions, they are imperative.

Support Requirements

There are many individual support items needed in any limited war type engagement. However, for our consideration only functional site support items shall be discussed.

Shelters are the structural items that are most frequently thought of and spoken about when expandable structures are mentioned. [1] Shelters are a natural candidate for expandable structure research. As they are necessary, a low volume/weight packaged expandable structure can provide a large inside clear structural volume necessary for a usable shelter. Other necessary support items are large enclosed storage areas needed for protection against climate and for camouflage. (See Figures 3 and 4) Pipelines are also necessary, either on a remote site for fuel and other oils transfer, or from ship to shore transfer or the possible transfer of fuel from control base to remote base. [2] Protection bunkers, fortifications and weather protection structures for ready arms are also needed. Hazardous or special fuel storage facilities are needed away from the main base areas, and expandable structures such as large storage tanks are necessary. Special purpose base operations structures such as photo reconnaissance, electronic warfare, or resupply airdrop structure package techniques (See Figure 5) are needed and often required in forward areas. [8] Expandable structures can assist in these areas by providing the special capability needed, and also reduce weight, volume, and logistic problems.

Figure 3

Figure 4

Figure 5

EXPANDABLE STRUCTURE TYPES:

There are many classifications of expandable type structures.[3] Many have listed these in various forms, but for consideration for limited war applications, see groupings as shown in Figure 2. Expandable structures generally offer the following advantages: low package volume to final volume, lightweight, easily transportable, may be rigidized or reused, low erection time, fewer required supporting personnel, generally have a high strength to weight ratio, and can be constructed to various shapes. The structures as listed in this grouping are: Inflateble, Unfurlable, Foamable, and Unfoldable.

Inflatable Structures

Inflatable structures are listed as follows: (a) those that are rigidized by inflating with a gas, thus catalyzing and rigidizing the flexible fabric by crosslinking, (b) those that are self rigidizing by other means, such as release of a subliming material which then rigidizes the system (plasticizer boil off) or (c) those that are dependent upon air or other gas for support and rigidity. All these types have advantages of low package weight and volume. The self rigidizing structures have good load carrying capacity without possibility of collapse due to air leakage, such as the air supported structures, but may not be reused as the air supported structure. Each may have use for various applications. Various materials may be used for these structures, such as encapsulated reactants,[1,4] airmat,[3] honeycomb [1,3,4] and other plastics and resins.

Unfurlable Structures

These are structures that expand by various means, such as telescoping and accordian type methods. These types may be elongated into final volume by pulling or utilizing other outside energy source. Compressed structures are those that unfurl or expand when either a gas, liquid, or force, causes the structure to expand, as a compressed sponge type material does when absorbing water. The final shape and volume may become rigid after this process. Stretch type structures are those that are stable when static but when a force is applied may stretch to a final volume. This may be an "Accordian type" structure or a stretch type cover over an expanding rib structure system. Such structures may also be stretched beyond the material yield strength for final shape stability. These structures may be built of various plastics, including memory plastics,[5] rubber type materials, metal structural systems,[3] and other resins and foams.

INFLATABLE • GAS INFLATED • SELF RIGIDIZING • AIR SUPPORTED MATERIALS: HONEYCOMB, SKIN FABRICS, PLASTICS, VARIOUS RESINS	***UNFURLABLE*** • TELESCOPE TYPE STRUCTURES • COMPRESSED STRUCTURES • STRETCH-TYPE STRUCTURES MATERIALS: PLASTICS, RUBBER, PANELS, METAL FOAMS, VARIOUS RESINS
FOAMABLE • SPRAY MOLD FOR RIGIDITY • ENCAPSULATED FOAMS • IMPREGNATED COMPONENT REACTION MATERIALS: URETHANES, EPOXIES, OTHER RESINS, & PLASTICS	***UNFOLDABLE*** • MODULAR TYPES STRUCTURES • VARIOUS GEOMETRIC SHAPES DOUBLE/SINGLE CURVED FOLDED PLATE MATERIALS: VARIOUS PANEL STRUCTURES, PLASTICS, WOOD, METAL

Figure 2

Foamable Structures

These are structures that may be fabricated by utilizing such reactants as polyurethane foaming agents. Many investigations have been conducted for utilizing foams for structural applications.[6,7,8] The latest investigative research is attempting to encapsulate these reactants[1,3,6] for application to expandable structures. The encapsulated inert reactants would be applied to an inflatable mold, and then compactly packaged. Upon inflation, the capsules would be ruptured, the structure would then be "foamed" and rigidized. The inflatable mold could be reused. Other research has been accomplished to utilize spray urethane foam for shelter formation.[9] "One shot" foaming systems have also been investigated for urethane structural applications.[7]

Other means of foaming structures may be by confining an inert epoxy or urethane foam reactant between two flexible skins, which rigidize upon foaming, or by impregnating or placing a resin, plastic or other component on or into a fabric which, upon addition of water, heat,[10] or other gas or liquid chemicals, foam and rigidize the structure.

Unfoldable Structures

These are the classical modular or other basic type shelters or structures which are pre-fabricated and erected at a site. They may come in various geometric shapes, incorporating designs such as plastic folded plate structures,[11] geodesic domes, and double curved designed structures. These normally would require the longest set up or erection time of all the types described.

EXPANDABLE STRUCTURES APPLICATIONS

Figure six (See Fig. 6) summarizes those expandable structure applications that can be used for limited war. Figure six (6) integrates the support requirements of figure one (1) and the expandable structure types of figure two (2), and shows applications and some distinct usages or advantages for each system previously described. As this figure summarizes and integrates information previously described, it will not be discussed in detail.

CURRENT SHELTER SYSTEMS

As outlined, expandable structures offer many distinct advantages over conventional structures, especially shelters, and of these numerous advantages, erection time of current shelter systems offers a good comparison point. (See Figures 7 & 8)

ENVIRONMENT USAGE

- NOT LIMITED TO ANY PARTICULAR ENVIRONMENT. EXTREMES MAY CAUSE VARIATIONS OF DESIGN

SUPPORT REQUIREMENTS

MAY BE USED FOR:

- SHELTERS-INFLATABLE OR RIGIDIZABLE
- STORAGE-FOR FUEL,- AGAINST CLIMATE
- PIPELINES- EXPANDABLE PIPE
- PROTECTION-CAMONFLAGE-WEATHER
- BASE OPERATIONS BLDGS.

LOGISTICS SUITABILITY

- EXPANDABLE= SMALL PACKAGE TO FINAL ERECTED VOLUME
- REUSABILITY

EASY TO HANDLE-NOT MANY COMPONENTS
AIR TRANSPORTABLE- LIGHTWEIGHT
QUICK USABILITY-LOW MANHOURS REQUIRED FOR FINAL STRUCTURE

CRITICAL SUPPORT AREAS

- ESPECIALLY SUITED FOR CONDITIONS OF:
  - REMOTE
  - FRONT LINE
  - EMERGENCY
- SOME MAY BE REUSED
- MAY BE USED TO ADVANTAGE IN UNDER DEVELOPED COUNTRIES

Figure 6

Figure 7

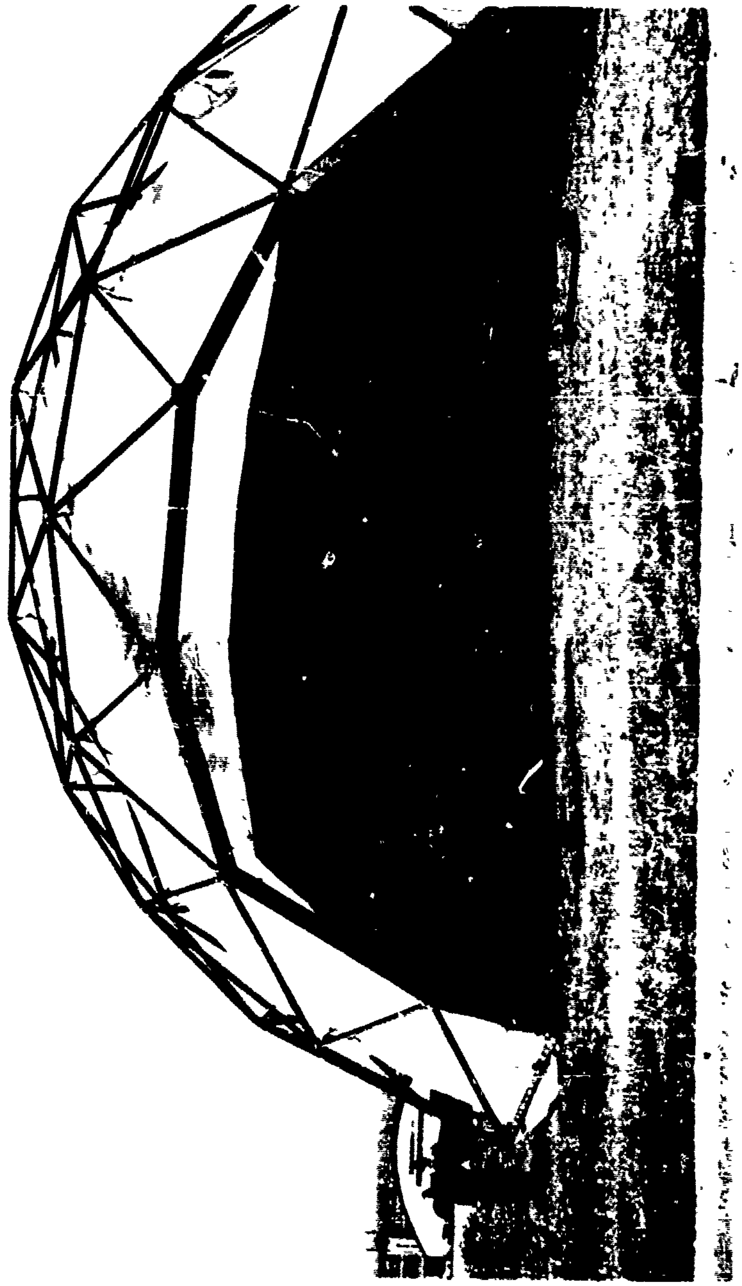

Figure 8

Figure 9 shows the various erection times of current design shelters available for use in remote front line areas today.[12,13]

As can be seen by this table, required erection times of buildings 20' x 48' in size are 40 to 188 manhours, and shipping weight and cubage vary considerably. Larger buildings show even greater variance.

A wooden, straight sided, 20 x 48 foot building had a total shipping weight of 11,250 pounds, a total shipping cubage of 288.5 cubic feet, and required 146.5 manhours to erect. Another sandwich panel building, modular type, decreased the erection time, but increased the shipping weight and cubage. This 20 x 48 type structure had a total shipping weight of 13,645 pounds, a 1406 shipping cubage, but took only 40.5 manhours to erect.

Arch type buildings generally utilize less cubage, total weight and erection time, but the reductions are not significant. A 20 x 48 arch type metal barracks shipping weight was 9,759 pounds, cubage equalled 162.5 cubic fe ., and erection time was 97.1 manhours. Larger structures shipping weight, cubage, and erection times are given in table one (1).

In comparison to these figures, in an expandable structure demonstration to be given under AF Contract AF33(657)-10409,[4] an expandable cylinder structure of 7 feet in diameter (final volume equal to approximately 400 cu. ft.) had a shipping weight of 100 pounds, occupies only 2 cubic feet of shipping volume, and upon experiment initiation is programmed to rigidize in less than an hour. This experiment dramatically points out the great advantage to be gained by use of terrestrial expandable structures. A 20' x 48' expandable structure would have an estimated shipping weight equal to 2,400 pounds and cubage of 50 cubic feet and an erection time of 12 manhours.

EXPANDABLE SHELTER ASSETS

These previous figures establish that, if an expandable structure shelter could reduce erection time one third, it would be of tremendous value. To reduce the shipping weight, cubage volume, and erection time is a much needed advancement to the state-of-the-art. Expandable structures are capable of doing this! The very nature of an expandable structure system allows these three logistic items to be reduced proportionally. Their value to a logistic system servicing a remote or forward area would be extraordinary. The simplicity of design and operation of the expandable structure system would create an even greater asset to the total usefulness of such a system. Therefore, expandable structures do, and must have a future place in any limited war support actions.

BUILDING COMPARISON DATA

DESCRIPTION	SHIPPING WT. (LBS)	SHIPPING CUBAGE (CU.FT.)	ERECTION MANHOURS
20' X 48'			
STRAIGHT-SIDE	9,042	138.5	95.0
	10,010	230.4	188.3
	11,250	288.5	146.5
ARCH TYPE	9,759	162.5	97.1
	10,670	359.1	96.6
	9,568	225.5	130.5
40' X 100'			
STRAIGHT-SIDE	25,984	509.5	508.7
	30,000	896.0	649.3
ARCH TYPE	29,204	429.0	627.3
	19,425	274.0	323.0
GEODESIC DOME (55' DIAMETER)	2,170	167.0	63.0

Figure 9

EXPANDABLE STRUCTURE RESEARCH

The applications and advantages for expandable structures have been very carefully established. Their usefulness in remote and front line areas for limited war actions have been highlighted. The limiting factor that now prevents this technology to be applied to this application of limited war is that of further research.

More research must be accomplished to determine and design specific usable concepts for expandable structures. This paper has stressed the requirements and applications, but concepts must be formulated, and design approaches established. Some research has been accomplished for specific purposes of expandable structure types,[14] but more must be completed before acceptance of the useful expandable structures.

In some instances, this will mean qualification proof testing to rid the old fashioned "2 by 4, sixteen inch studs on center" frame of mind that has entrenched itself in unprogressive, archaic codes of cities and armed services alike. We must move forward by presenting factual, undisputed data for acceptance of this technology. Limited war applications of expandable structures offers us this opportunity.

Conclusions:

1. Expandable structure technology is feasible.

2. Expandable structures offer definite advantages for limited war usage.

3. Further expandable structure research for limited war applications is necessary.

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SPACE MAINTENANCE

HANGAR CONCEPTS

BY

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SPACE MAINTENANCE HANGAR CONCEPTS

INTRODUCTION

Providing man with a maintainable system becomes a major problem area as manned space flights become more extended in time and through space. A decision has to be made early in the system design phase as to how the system will be maintained; such as, add redundancy, abort the mission, repair or replace a faulty component. If man wants to obtain the capability to perform long space flights, he must resort to the repair and replace philosophy. The Support Techniques Division of the Aero Propulsion Laboratory has for the past three years been engaged in an exploratory development program to obtain the capability to assemble, maintain, repair and replace components of a space vehicle in a space environment. The program has been divided into four categories: (1) Extra vehicular activities (method of propelling man between spacecrafts to perform maintenance, repair, rescue and space station assembly), (2) Tools, fasteners and attachment techniques, (3) Maintenance philosophies and concepts, (4) Protection for the man while outside the vehicle. This paper is concerned with Category 4 and will specifically discuss space maintenance hangar concepts.

PURPOSE: The primary purpose of this paper is to present a discussion on the justification, requirements, and possible structural concepts for a space maintenance hangar. An objective of this discussion is to stimulate other investigators to consider future research in this area.

SCOPE: This paper will cover the general results of the Air Force space maintenance programs, requirements for a maintenance hangar, design concepts for a maintenance hangar criteria, and recommendations for future work in this area. It is not within the scope of this paper to present a detailed design for a specific space hangar concept.

For a discussion of hangar concepts for maintenance application in a cosmic environment to have any real meaning, one must have some knowledge of the background establishing the requirement for this hangar. Therefore, the approach used in this paper is to first discuss the efforts and investigation completed to date that have been directed toward obtaining the capability to maintain, assemble, replace/repair systems in a space environment.

SPACE MAINTENANCE BACKGROUND

Space Maintenance is defined as the ability to perform maintenance on a space vehicle sometime between launch and recovery. This is a simple capability to state in words, but a difficult one to achieve in space, due to the hazards and problems imposed by the environment. Factors such as lack of spares, no maintenance depot, cramped quarters, high vacuum, radiation, light contrasts, suit mobility, are some of the problems to be solved in achieving a maintenance capability. Why then pursue it at all? The following discussion is an attempt toward answering this question.

As man orbits the earth for longer and longer periods of time or travels to the moon or other planets, the ability to return him in an emergency becomes more and more impractical. Therefore, the main reason for in-space maintenance is survival of man himself. The key to successful long term space flight is a 100% reliable system which is currently impossible. This reasoning stems from the relation: $R_s = R_i$ where R_s is overall system reliability, R_i is individual component reliability and is the number of elements. From this relation, it can be seen that with more complex systems having more components, the probability of success decreases. This being the problem, then how can reliability be increased? There are three basic techniques which are as follows:

1. Improve individual component reliability

2. Use redundant parts

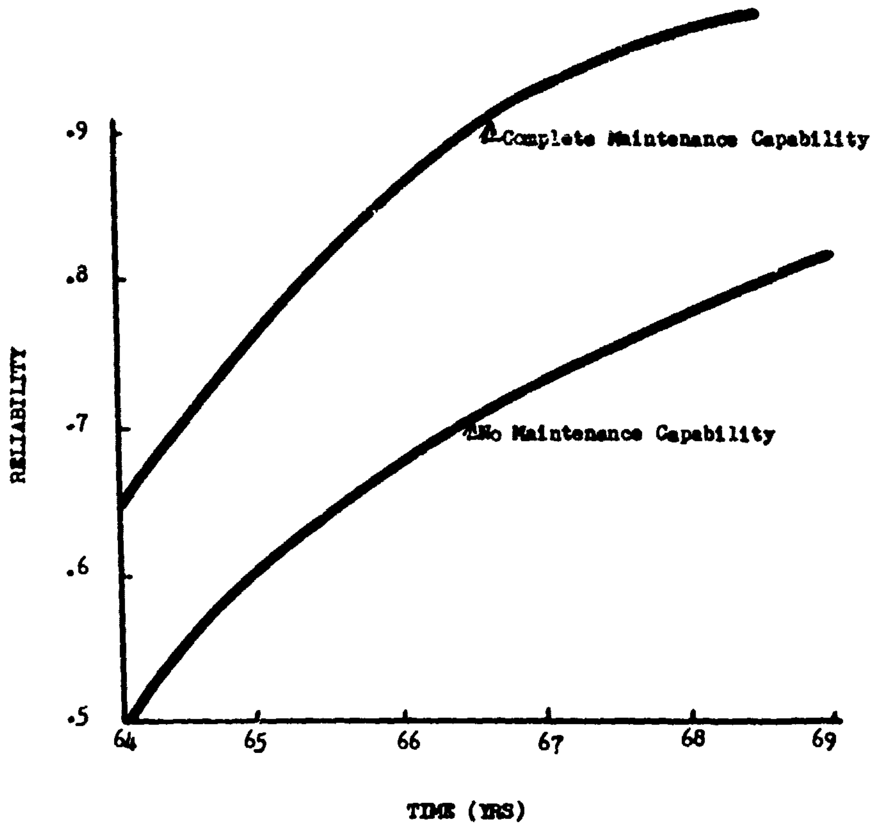
3. Provide repair capability

The first method is self explanatory. The space system designer and builder places emphasis on product design, quality control, and testing to improve the reliability of each component. This method is slow and progress becomes asympototic as the reliabilities approach 100%. In Figure 1, Curve A shows the predicted increase in system reliability due to technological advances on individual components. This improvement cycle must be accomplished on each new weapon system, and because each new system has new components this process is accomplished only after considerable time has elapsed.

The second method is redundant components in parallel. The mathematical relation expressing redundancy is:

where mi is the number of components in parallel. A high degree of reliability can be achieved by adding more and more components in parallel. Other factors enter the picture to prevent all but a limited use of this method. These limiting factors are weight and cost. Booster technology and

FIGURE 1



SPACE MAINTENANCE
COMPARISON

capabilities have been governing factors on physical size and weight of spacecraft. This has resulted in miniaturization and micro-miniaturization of components in order to give man the necessary equipment to perform his mission. Redundancy has limited to electronic components, and to some critical control functions. This was illustrated in Project Mercury, where redundant attitude controls were provided. Therefore, trade-offs between redundancy, costs, and weights are necessary.

The third method is to provide a repair capability. This has the potential of achieving full redundance without the cost and weight. It also involves man to some degree. There are three levels at which man participates:

1. System Monitor

2. Interior Maintenance

3. Complete Maintenance

As a system monitor the Astronaut will monitor all performance sensors and detect system degradation or malfunctions. He serves as system interrogator and mentally relates the equipment status to determine what action is required to correct malfunctions. Normally, his "on-the-spot" judgment plus ground tracking data jointly establishes the course of action whether it be to deorbit or to continue mission with a degradated system. The only repair capability provided in this method is switching in redundant parts.

Since the study by Bell Aerosystems Company (Ref. 1) indicates that 60% of all malfunctions of a manned system can be reached from within the spacecraft, it is then possible to provide the Astronaut with a limited maintenance capability. Such a capability would be limited to electronic, life support, and environmental control systems. Due to the severe volumetric limitations of current spacecraft, this would not provide a sufficient increase in maintenance capability over the first level of participation to warrant consideration. However, with future spacecraft such as orbiting space stations, the Astronaut could accomplish many repairs from within the cabin.

The complete maintenance capability involves man exiting the spacecraft to perform extra vehicular repair tasks on his or other space vehicles. This concept would near the 100% reliability desired in spacecraft provided man has the capability to exit the craft with the tools and necessary spares to accomplish the repair.

Man uses his visual senses to survey the performance indicators, his mental capacity to interrogate this instrument to determine criticality of the malfunction and decide the course of action, and finally his physical strength and dexterity required to accomplish the repair. Leaving the spacecraft he enters a hazardous environment and will require the following:

1. Physiological protection against the environment.

2. Propulsive means to move him from place to place.

3. Tools and equipment to accomplish the tasks.

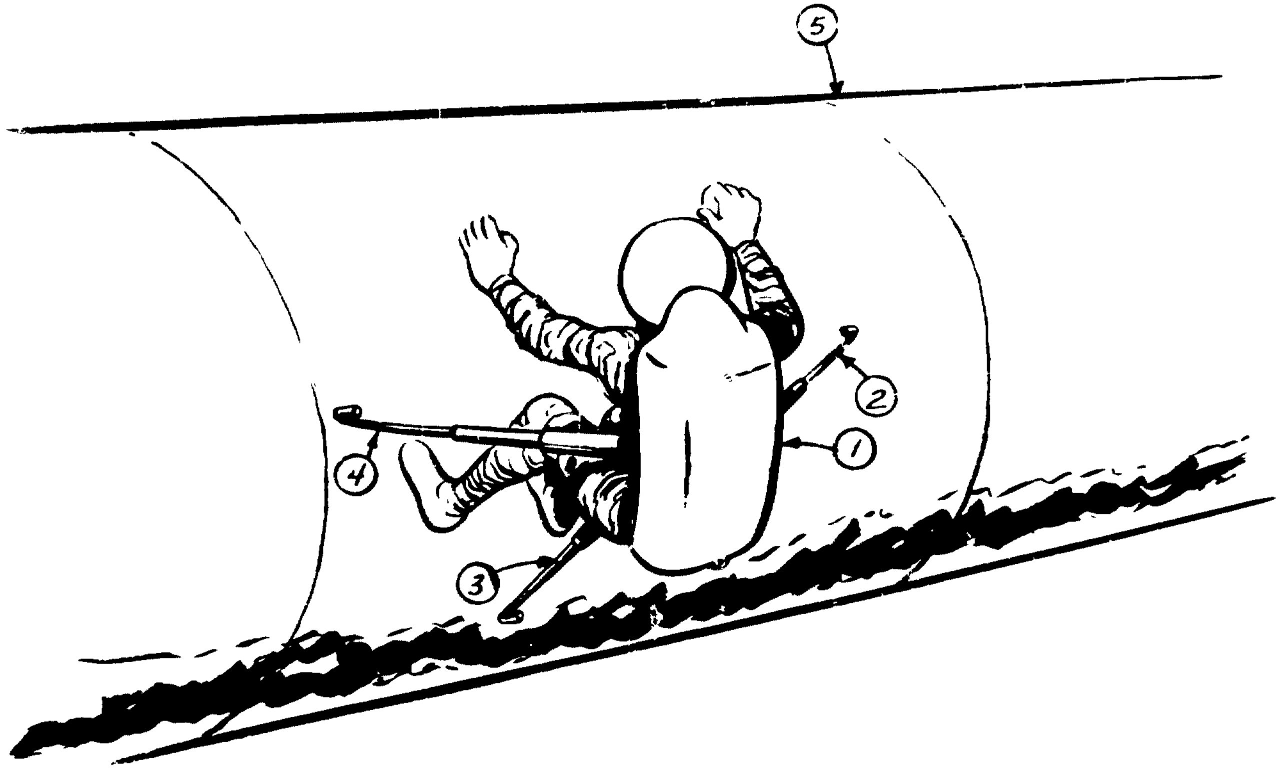
4. A preplanned maintainable vehicle.

This paper will discuss these factors in reverse order. First, a preplanned maintainable vehicle is one into which in-space maintenance has been designed. A study of potential spacecraft (Ref. 1) shows that space vehicles in being or in the planning stage are not designed for maintenance once the vehicle leaves the pad, except for a few redundant components in critical areas. This study shows that components which are likely to fail would require a major disassembly effort to gain access to the defective part. In addition, the fasteners utilized in most cases would be difficult to remove even in the normal environment. Efforts are being made by the Air Force Aero Propulsion Laboratory to establish the necessary criteria so that space vehicle designers can design their craft to be maintainable in space.

Second, the proper tools and equipment must be provided, and this effort be conducted concurrently with the design criteria and maintenance concepts. Today research and experimental testing in an actual weightless environment is being accomplished to develop low-torque multi-purpose tools (Fig. 2). As the Astronaut will be in weightlessness outside the spacecraft, he must attach himself to the worksite so that the low torque transmitted to him by the tools can be transmitted to the work site. One such technique is shown in Fig. 3. This method utilized three light weight telescoping tubes which can be extended to the spacecraft by the Astronaut. Pads would be located on the end of these tubes and would contain contact adhesives. Once these pads are pressed against the surface, they will instantly make a bond. The maintenance worker simply pulls a quick release pin and telescopes the tubes to a minimum package after completion of this mission.

The third requirement to leave the spacecraft is a propulsive device to move about the vehicle surface or to a second vehicle. The various techniques to accomplish this are discussed in (Ref. 2). This analysis (Ref. 3) shows that the Astronaut Maneuvering Unit is the best device today for providing this capability. The Astronaut Maneuvering Unit (AMU) (Fig. 4) is a self contained package containing propulsion, stabilization, life support and environmental control, and its own power supply. This package is back-mounted to keep the arms and legs free. It can be designed to provide ranges to 30,000 feet and a 4-hour life support capability. Short range AMU's could be designed to fit either the back or chest depending on which type and size spacecraft it can be carried. Typical variations are shown in Fig. 5.

Figure 2



ARTIST CONCEPT OF INSTANT
BONDING ATTACHMENT MECHANISMS IN USE
SKETCH #1

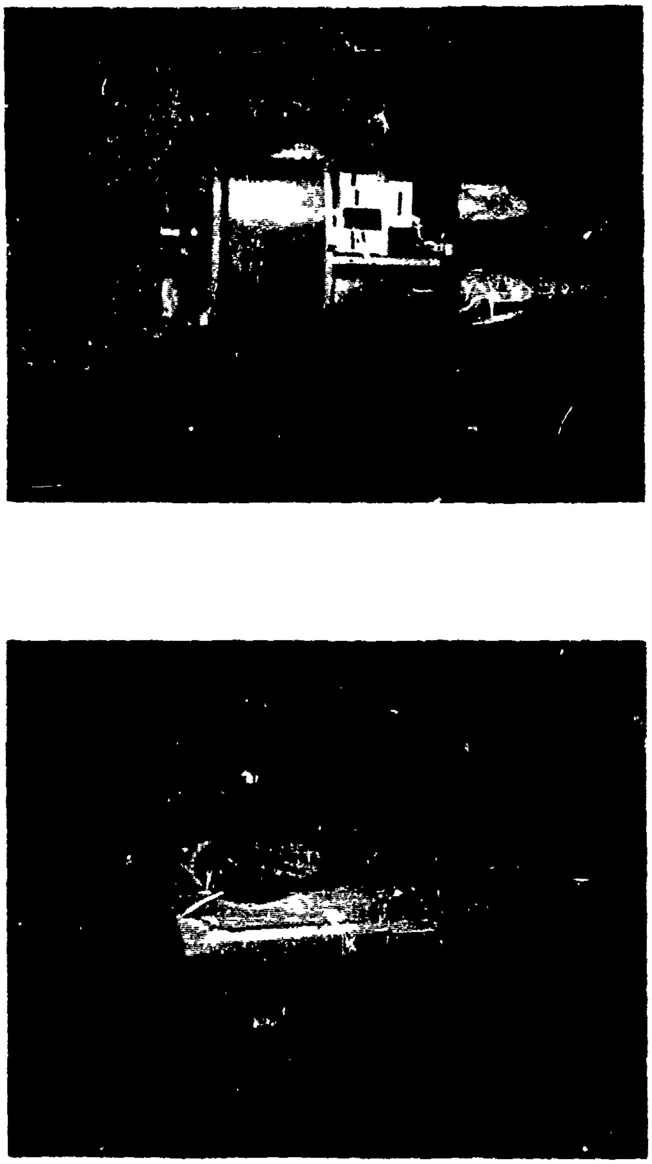


Figure 4

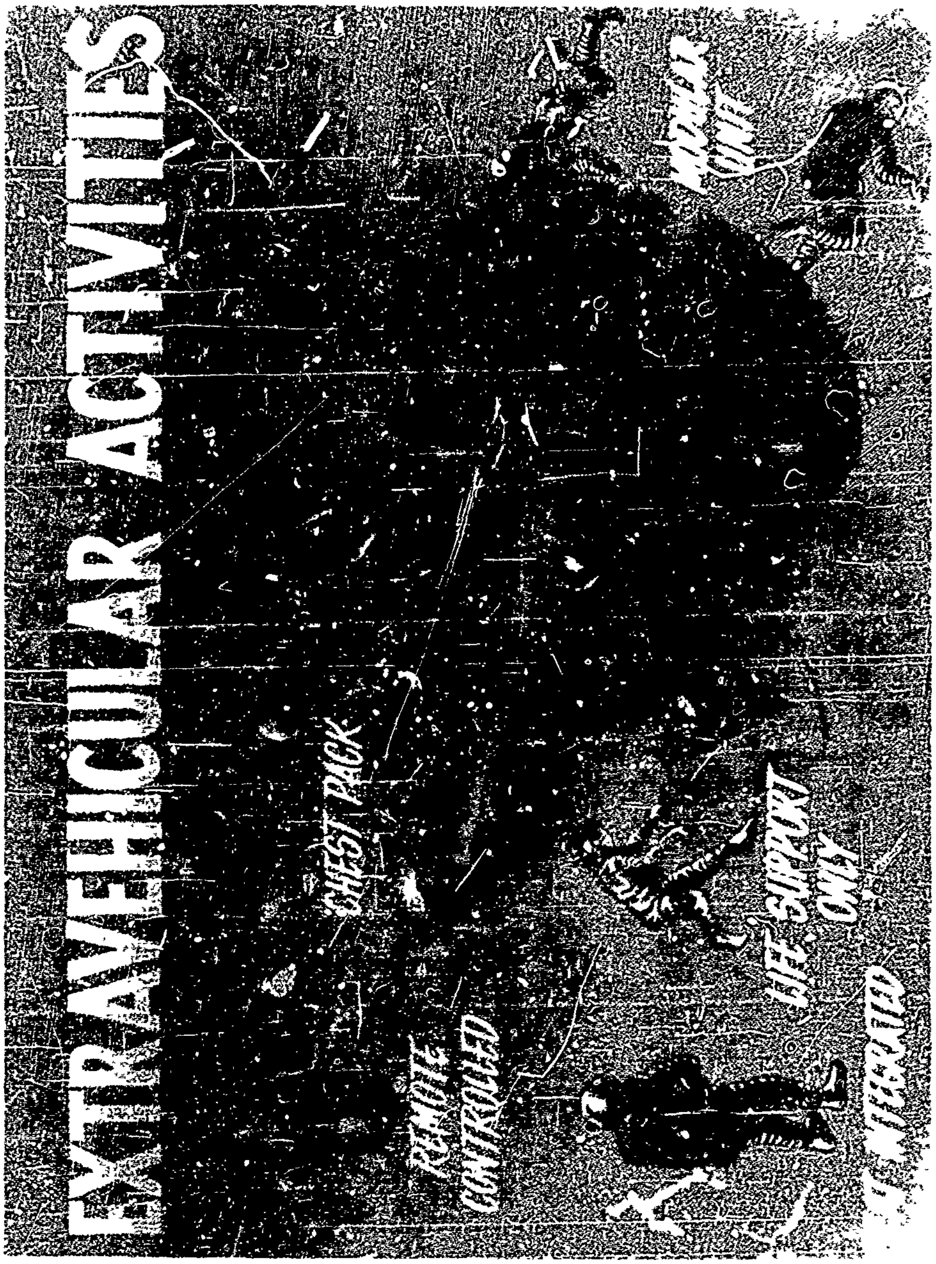

EXTRAVEHICULAR ACTIVITIES
CHEST PACK
REMOTE CONTROLLED
LIFE SUPPORT ONLY

Figure [illegible]

Finally, the fourth, but most important consideration for extra vehicular excursions is protection of man against the environments. There are three categories of protection which can be afforded to the man. These are:

1. Space Suit

2. Space Capsule

3. Space Hangar

The primary purpose for this paper is the latter, a maintenance hangar, although space suits will also be discussed. Briefly, these three differ by the length man is able to stay outside the spacecraft. Research on space suits over the next five years will give man the capability for at least a two hour excursion outside his vehicle and may with the technological breakthroughs attain four hours. The space capsule (Fig. 6) is a hard shell enclosing the man and giving better protection. The capsule can increase the capability to 8-12 hours; however, it has two distinct disadvantages. One is increased weight. The AMU (150#) plus suit and man (190#) weighs about 340# while the capsule would be 700-800#. The second problem is that man must now use manipulators for hands and will increase the time to accomplish a task by about 10 times. The space hangar concept combines the flexibility of a suit with a protective shell to provide the most effective means of performing space maintenance.

Several hangar concepts will now be presented and the requirements used to establish these concepts will be discussed later.

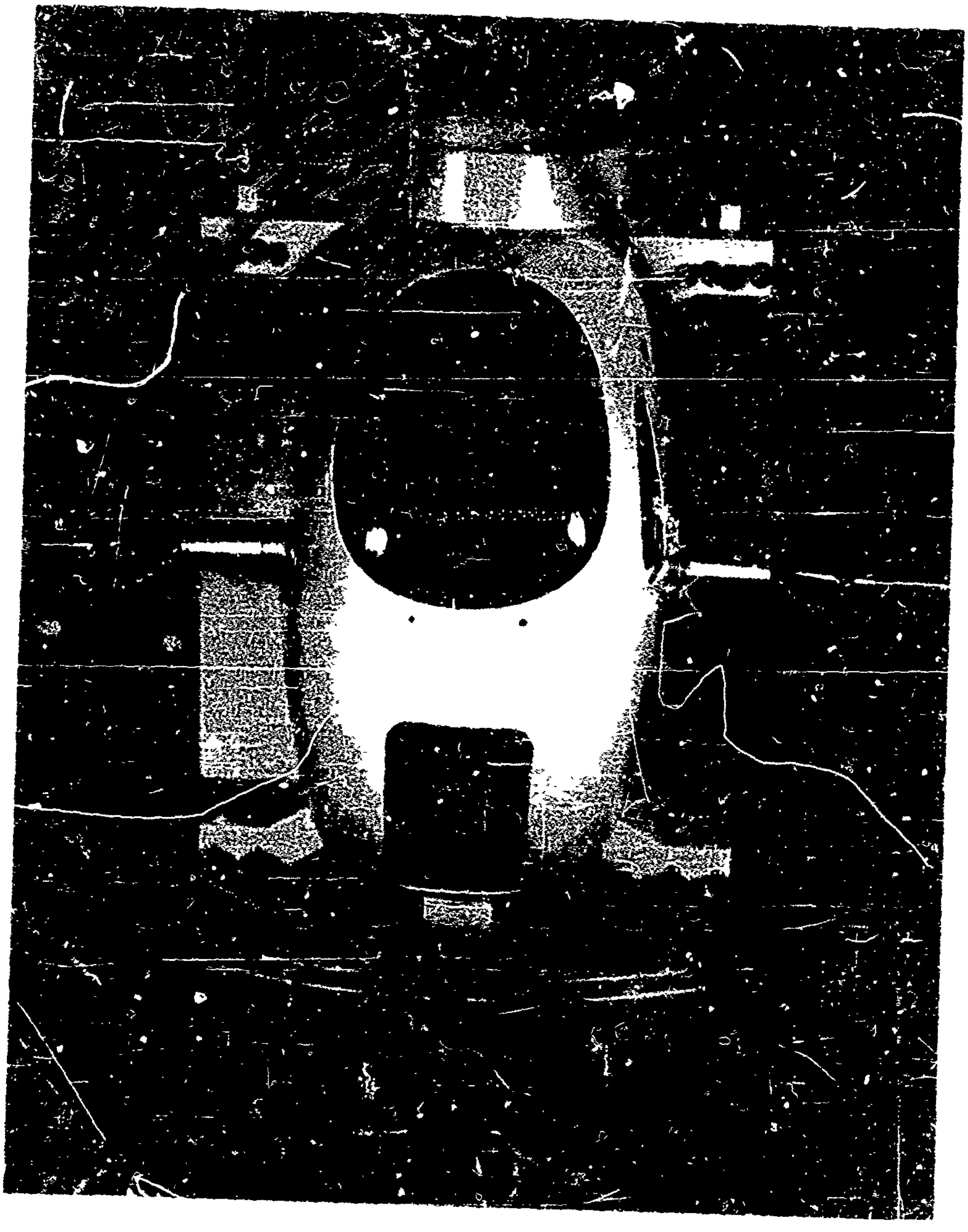


Figure 6

General Concept:

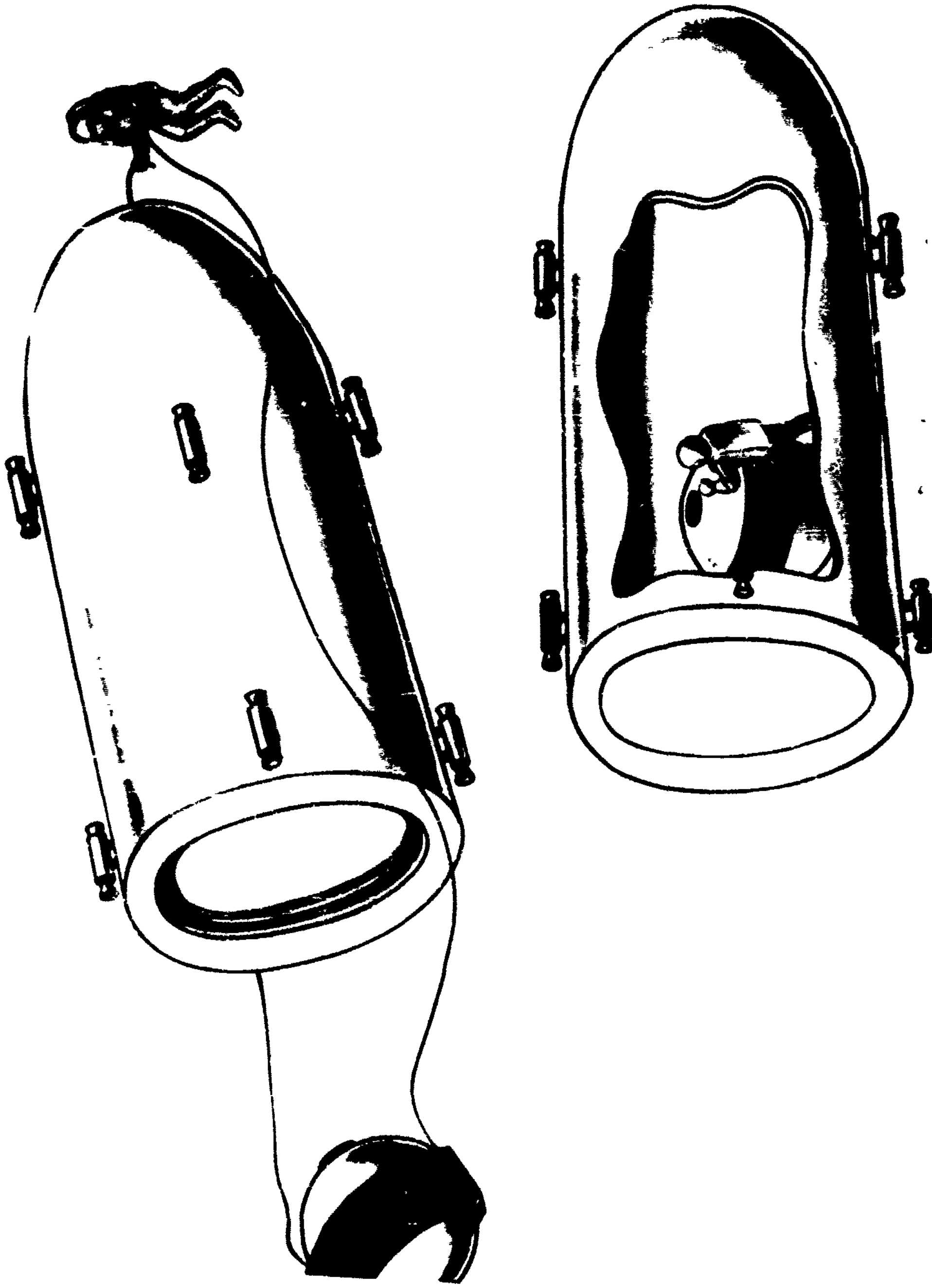
The simplest type of s
and operational point of vi
hangar, as shown in Figure
materials now in use. The
struction, with exterior sh
frames, honeycomb, or rigid
(Figure 7) would be a simpl
airlock would be at the opp
swung towards the inside of
ing problem of an internall
the hangar door or airlock
the overall vehicle length
instance, if a cylindrical
of maximum dimensions 20 ft
need a hangar of at least 5

Materials:

As previously mentioned
which would be available f
struction of a rigid mainte
of design would be a double
inch cell size aluminum hon
rigid urethane form. Rigid
airlock opening and possibl
The double wall constructi
protection, while the hone
high strenght-to-weight rat
would minimize extensive da
fragmenting of the exterior
penetrations. One of the m
hangar concepts are seals f
alleviate this problem is t
will be relatively small.
with a higher leak rate tha
application.

Proposed Operation:

The vehicle may have it
Astronaut Maneuvering Unit
over a disabled vehicle. C
objective, the space mainte
the airlock. The hangar wi
worker's space suit pressure.
power supply system. Chemi
collector could be used to



A Rigid Hangar concept

Figure 7

Advantages:

This type of space hangar has several advantages; considerable knowledge on design procedures exist for a rigid double wall cylinder, utilized conventional well known materials, and completely assembled can be checked out on ground. Each of these advantages has considerable merit; however, this type of hangar concept has several significant disadvantages.

Disadvantages:

One main disadvantage of this system is that it has definite size limitations. In other words, the maximum hangar diameter must be compatible with its launch booster. It appears unrealistic considering the size of even the largest booster to attempt to put a hangar of larger than twenty-five feet in diameter into orbit. Another point of interest is that a large hengar would probably require the largest booster in the inventory from a size standpoint, but from a weight standpoint would utilize a significantly smaller booster. It is relaized that the inside of such a hangar could possibly be utilized to transport secondary cargo into orbit. Most likely this type of hangar has application for maintenance on small unmanned satellite systems.

TELESCOPING SPACE MAINTENANCE HANGAR

General Concept:

A cursory review of previous discussion on the rigid cylinder hangar shows that the only practical way to place large size maintenance hangars into space requires the use of an expandable structure. This conclusion is based on the fact that boosters have a definite size limitation. There are many basic types of expandable structures; however, this paper will discuss only telescoping, inflatable, and chemically rigidized structures. The first structural concept to be discussed will be the telescoping concept.

A telescoping maintenance hangar, as shown in Figure 8,would be composed of a number of sliding cylinders. Each cylinder section would be slightly larger than its adjacent section on one side only. A telescoping hangar of four (4) sections in length could be sent into orbit in a launch configuration of one quarter its total length plus the height of a domed end cap. It is significant to note that the launch package must still be of the same diameter as the fully expanded hangar. Nevertheless, a telescoping hangar has some applications where it is desirous to place an extremely long hangar into orbit as opposed to its diameter.

The hangar, as shown in Figure 8,would have one hemispherical end in which would be located life support, power generating equipment, fuel for reaction jets, etc.; and at the other end of the hangar would be the airlock door. Again, this door would by necessity be swung into the hangar. A flexible bladder could be fastened to the inside of the structure to minimize air leakage.

Materials:

The materials that the authors would proposed for a telescoping space maintenance hangar, are similar to the rigid cylinder materials. These are two sheets of aluminum alloy separated by aluminum honeycomb filled with a urethane foam attached to forged aluminum alloy frames located at each end of telescoping sections. A metallic bar X seal could be utilized to minimize leaks and expedite deployment. This bladder could be a woven organic fabric impregnated with a flexible urethane plastic film.

Proposed Operation:

The space maintenance hangar, after being boosted into orbit, could either be automatically deployed or deployed semi-automatically, utilizing the capability of the space maintenance astronaut.* Once the hangar has been extended, the reaction jets could be operated by the maintenance worker in such a manner as to capture the disabled satellite inside the hangar. Then the airlock door

*Deployment of a telescoping structure may be accomplished by either a mechanical action or internal pressurization.

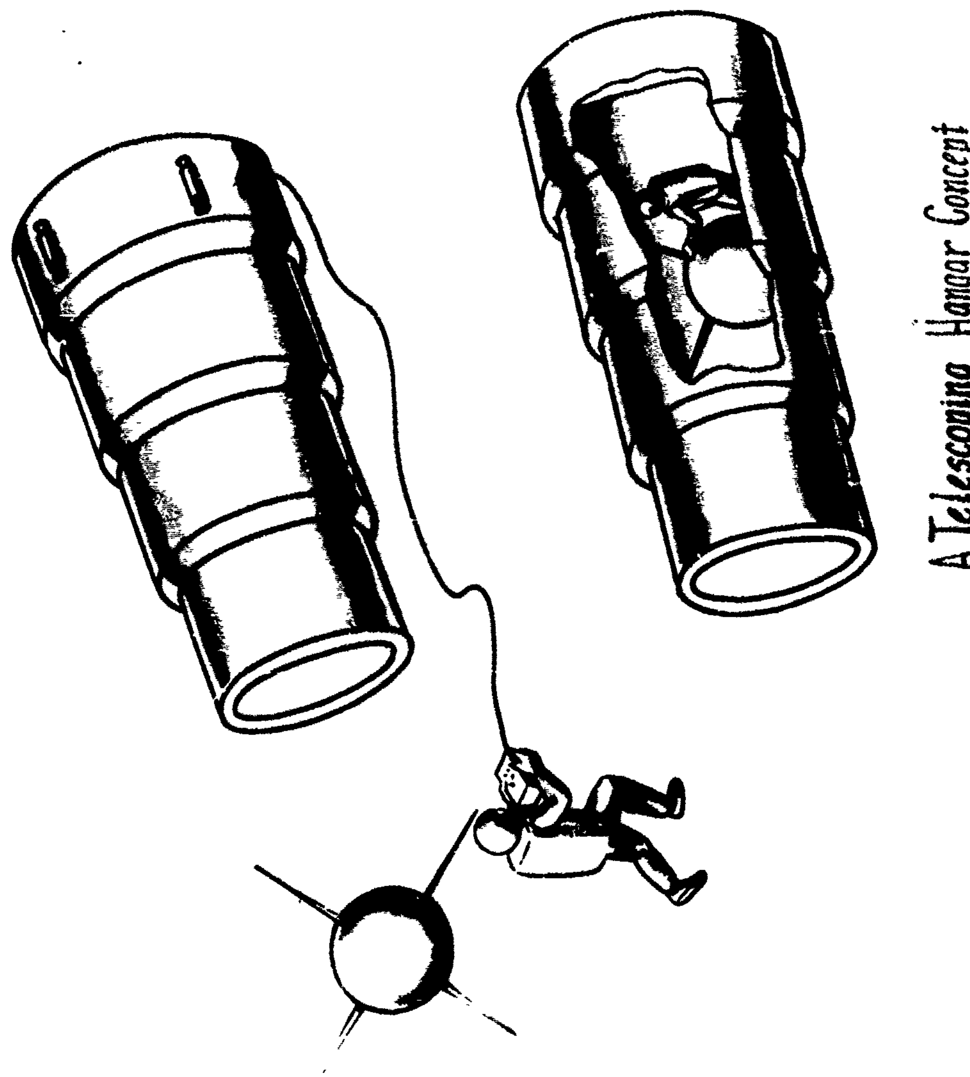


Figure 8

would be shut by the repair worker and the inside of the hangar pressurized to just equal the inflation pressure of the astronaut's space suit. This zero pressure differential, as will be discussed later, would significantly decrease the time to perform specific repair tasks. Needless to say, the maneuvering of the hangar over the target vehicle could be accomplished by use of the astronaut maneuvering unit or the astrotug.

Advantages:

A telescoping hangar has two main advantages: the capability of significantly reducing the overall length of the hangar during launch and the use of conventional rigid materials.

Disadvantages:

There are several significant disadvantages associated with a telescoping structure. A study by the Martin Company under Air Force Contract No. points out that a telescoping structure designed to do a specific mission is about 1.4 times (ref. A) as heavy as a rigid conventional cylinder designed for the same mission. This is a significant disadvantage relative to booster payload capability. Although, as previously pointed out, a designer can significantly reduce the launch package length via a telescoping structure; a major disadvantage is that the diameter of the package cannot normally be reduced. The diameter is usually the most critical launch package dimension. Finally, since there are considerably more joints in a telescoping structure, as compared to a rigid structure, the leakage loss for this type of structure would probably be higher.

INFLATABLE MAINTENANCE HANGARS

General Concept:

The next structural concept to be discussed is most familiar to expandable structure investigators, that is the inflatable structure. Inflatable structures have made significant progress through the passive communication satellite program of NASA.

An inflatable hangar, as shown in Figure 9, is similar to the shape of the rigid cylinder hangar. The inflatable cylindrical envelope would have a number of small inflatable tubes running parallel and perpendicular to the longitudinal axis of the hangar. These tubes when initially inflated would cause the hangar to fully deploy.

Materials:

The inflatable structural envelope would be fabricated from a woven rayon, dacron, or fortisan fiber impregnated with a flexible urethane, polyethylene, or other flexible elastomer. The tubes which are inflated for deployment would be of the same material as the envelope. The exterior of this inflatable structure should be covered with about a quarter to three eights inch of a flexible urethane foam for micro-meteoroid protection. Although it would add to the automatic deployment of the hangar a second materials concept which merits definite consideration for this type of application is airmat or a double walled inflatable drop thread structure.

Proposed Operation:

The inflatable space maintenance hangar, after being boosted into orbit, would be expanded by a combination of inflating the small diameter tubes shown in Figure 9, and utilizing the potential energy stored in the compressed flexible form. Once the structure has been deployed, the hangar could be towed or pushed over the derelict vehicle. When the disabled satellite is positioned inside the hanger structure, a torus would be inflated at the airlock end. This torroid will act as a frame for the airlock door. This torus would be inflated to a high inflation pressure after which an inflatable tapered plug would be placed in the torus. This plug would be placed into the torus from inside the hangar by a maintenance worker and would also be inflated to a high pressure. The hangar envelope could then be pressurized to the level of the astronaut's space suit pressure. This pressurization of the hangar would merely tend to drive the tapered plug more tightly into the torus, thus, creating a very good seel.

Advantages:

The inflatable maintenance hangar offers several significant advantages. This type of structure can provide the minimal launch package volume in all directions. This type of structure has a very high strength-to-weight-ratio. The use of the tapered plug airlock door could eliminate the hinged door concept which in turn could significantly shorten the required hangar lengths.

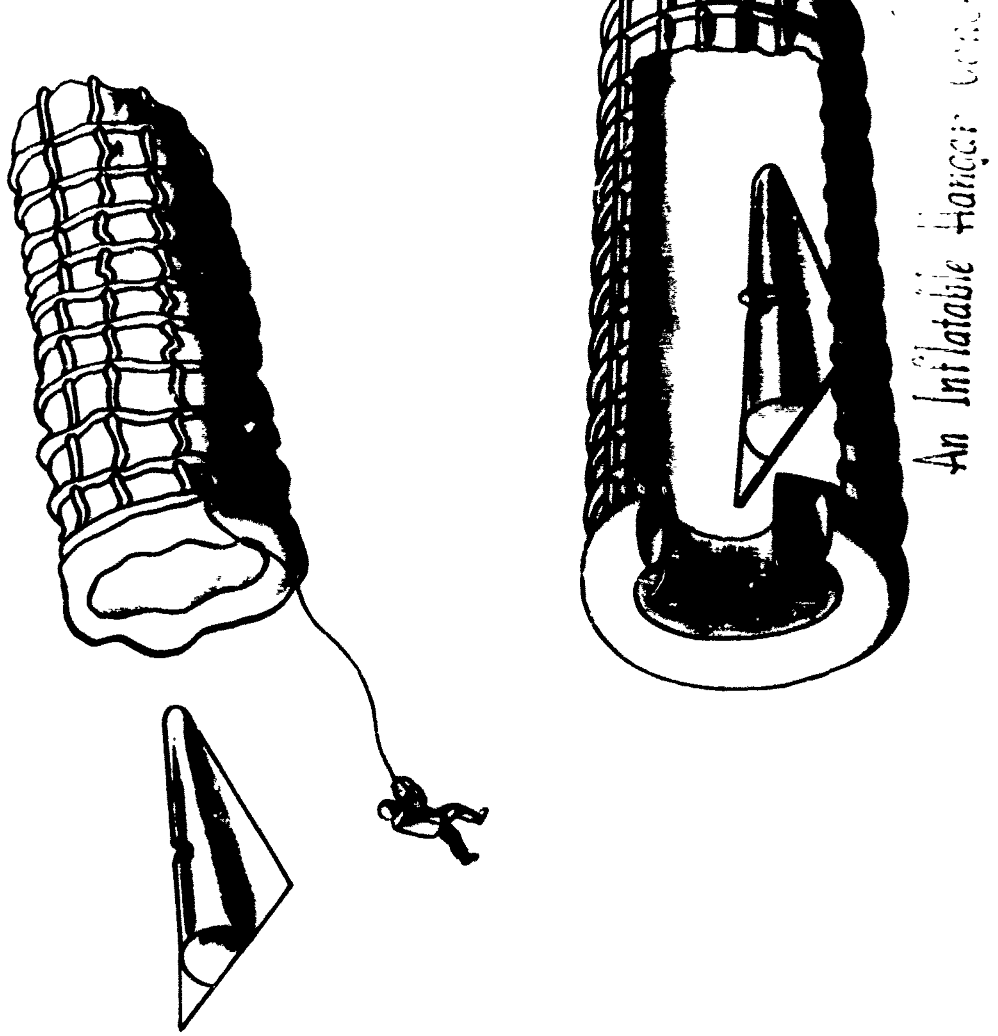


Figure 9

Disadvantages:

An inflatable structure has several disadvantages. Puncture can cause a loss of structural integrity. The plasticizer which causes a plastic to be flexible can migrate rapidly in a space environment and cause the flexible to become brittle.

CHEMICALLY RIGIDIZED SPACE MAINTENANCE HANGARS

General Concept:

The final type of maintenance hangar structure to be discussed is a chemically rigidized expandable structure. A chemically rigidized structure may be a foamed-in-place, rigidized membrane, or an expandable self rigidizing honeycomb. In the technical paper, "Expandable Structures for Aerospace Applications", it was concluded that the optimum rigid structure on a strength-to-weight ratio would be an expandable honeycomb. Therefore, in the category of chemically rigidized hangars, this paper will consider only an expandable honeycomb structure. This concept utilizes a woven fiberglass core pre-impregnated with a urethane resin or epoxy resin, semi cured to a flexible state. This woven impregnated core can be folded and packaged into a compact volume for launching into space. After arrival at the final destination, the structure is deployed and the honeycomb cell spaced inflated with a vaporized catalysis after which complete rigidization occurs. This type of hangar is shown in Figure 10.

Materials:

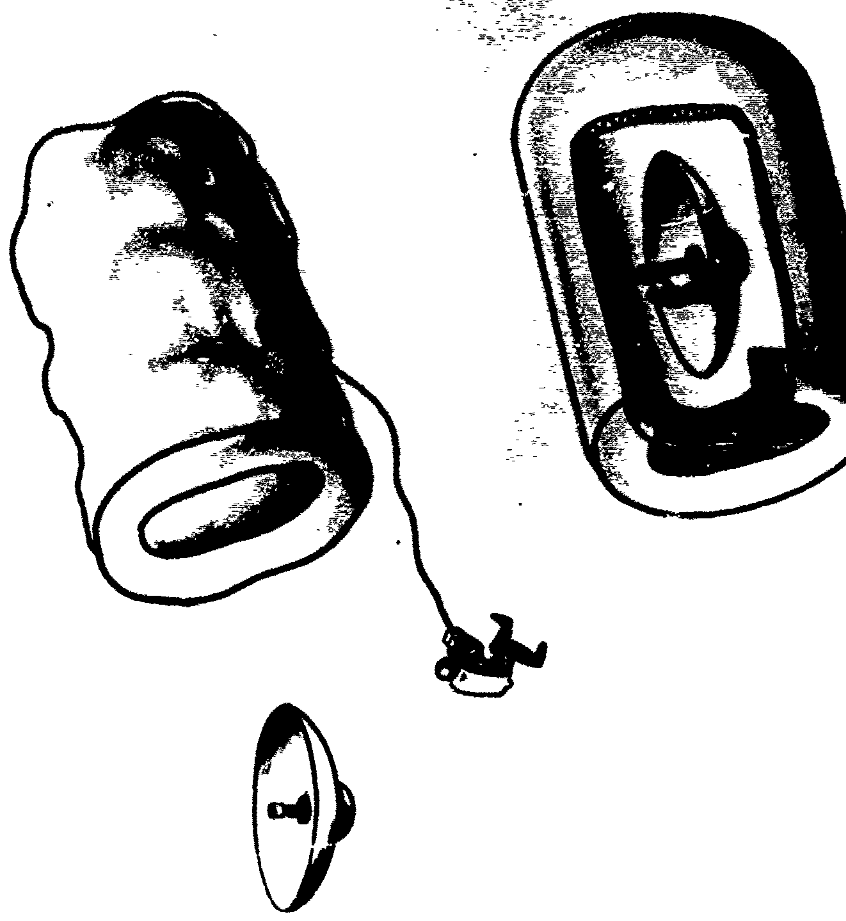
The optimum core configuration is probably a corrugated type of core where the corrugations run circumferentially around the hangar. It is realized that this type of core is not a true honeycomb; however, the circumferential stress is approximately twice the longitudinal stress in a cylindrical body. This unequal stress relationship can utilize the anisotropic properties of a corrugated core. An added benefit of a sandwich construction is the doubled wall effect which is considered desirable for micro-meteoroid protection. Currently, the optimum resin system for preimpregnating the core would be a urethane resin, cured to a B stage. The catalyst which would be utilized to inflate the core flutes and rigidize the structure would be water vapor with a small amount of an amine dispersed in it.

Proposed Operation:

This type of hangar, as shown in Figure D, would be boosted into orbit in a small launch package compared to the final expanded volume. A recent honeycomb 7 ft. dia. by 7 ft. high shelter was packaged into a two cubic foot package. After arrival in space, the structure could be expanded and rigidized automatically or semi-automatically utilizing the maintenance worker for performing some tasks. A large structure would be fully cured in about one hour; however, the maintenance worker could utilize the structure as soon as it was fully deployed. An inflatable torus and tapered plug could be utilized for an airlock, similarly to the inflatable hangar.

Advantages:

This type of structure has the following advantages: small package diameter and length, puncture of structure without loss of structural integrity.



An Expandable Honeycomb Hangar

Figure 10

Disadvantages:

The primary disadvantage of this concept is its inability to be repackaged after rigidization.

SPACE SUIT EFFECTS ON MAINTENANCE

General Concept:

After the previous discussion on space maintenance and structural hangar concepts, the reader may wonder why would a space maintenance hangar be required, especially when space suits are being developed to provide the same protection. Space suits, although they will afford man the necessary protection from the hostile environment of space, causes a severe degradation of man's ability to effectively perform maintenance tasks. Space suits also would never in themselves provide long repair/time capability nor provide adequate protection in other problem areas which will be discussed later.

Results of Maintenance Degradation Study:

The investigations conducted to date on the ability of man to perform maintenance in a full pressure suit have been rather limited in scope. However, enough data has been obtained to draw definite conclusions or trends on the negative aspects of a space suit relative to man's ability to perform space maintenance. One such investigation conducted by Lockheed Aircraft Corp. (Table 1) will be discussed in this paper.

Table 1 is a summary of the results of this particular investigation. It should be noted, for example that average time degradation obtained in this study was 236%, while the actual degradation times ranged from a minimum of 28% to a maximum of 450%. These tests were conducted by a subject performing certain specified tasks under the following conditions: in a shirt sleeve environment, in an unpressurized Mark IV suit, and finally in a fully pressurized Mark IV suit with the times being recorded per task. It is realized that current and future research being conducted in the space suit area should significantly reduce these degradation percentages. In addition, the research being conducted by the Aero Propulsion Laboratory in the area of space tool design should also reduce these repair times. However, the maintenance tasks selected for this study were of a very simple and elementary type.

A pressurized maintenance hangar would virtually provide a maintenance worker in a pressurized suit the capability of performing maintenance tasks essentially in an unpressurized environment. Therefore, utilizing the results in Table 1, it is easily seen that a significant reduction to maintenance time can be achieved through the utilization of a fully pressurized hangar in comparison to a pressurized space suit vs. a non-pressurized space suit.

TABLE I

Time to Complete Task

Test Number	Description of Task	Shirt Sleeve	Unpress. Suit	Fully Press. Suit	%Time Decrement	Remarks/Notes
1	Gross Arm Movement - Patty Cake Routine	-	-	-	-	No deterioration in time or quality. Add effort - more fatiguing - otherwise no effect.
2	Writing - copying a prepared paragraph	0.75	0.75	1.05	40%	Writing quality somewhat degraded - still legible. Quality could be improved with practice or by taking more time.
3	Putting nuts on studs. Installing 33-5/16 coarse threaded square nuts on studs - achieve full thread engagement	2.82	3.77	9.55	239%	Particularly affected by oversize gloves. I cc volume represents about smallest object which could be practically manipulated.
4	Removing radio rack panel Walking around obstructions.	2.00	4.14	11.00	450%	10-32 screws too small for practical operation with these gloves. Separate multiple test umbilical leads caused interference during walking (by snagging objects).
5	Removing and replacing amplifier tubes and cover.	0.62	0.63	0.83	34%	Very little decrement. These objects are large enough to handle.
6	Removing and replacing subpanel - 4 ea. 10-32 screws, nuts, washers.	3.20	5.23	11.00	240% on completed portion	Hardest task of all. Multiple small objects to manipulate, 10-32 screws, nuts, washers, panel, frame, screwdriver, and wrench (open end). Could not hold nuts to reinstall lower screws

Time to Complete Task

Test Number	Description of Task	Shirt Sleeve	Unpress. Suit	Fully Press. Suit	% Time Decrement	Remarks/Notes
6 (continued)						because of confined space and oversize gloves. Aborted unfinished. However, panel secured with 2 screws. Need for training pointed up by placing nuts in wrong place on table. Had to release panel and start over.
7	Changing power cables	1.8	2.20	6.	256%	Better selection of tools would help.
8	Removing and reinstalling cannon plug and safety wire.	2.45	2.98	6.[illegible]	169%	Wire hard to cut. Poor grip. Cannon plug knurled nut hard to turn against friction. On reinstallation, learned to push plug to eliminate high thread friction - training - quality of wiring.
9	Tying two bowline knots	0.41	0.62	1.58	285%	Clumsy gloves slowed. Otherwise no real problem.
10	Torso turning - look at cue card, turn 180° to set dial needle like card reading.	0.39	0.39	0.50	28%	Virtually no time decrement in basic task. Added time primarily caused by glove clumsiness in operating needle.

NOTE: 236% average time decrement for all tasks could be improved to 100% by use of proper tools and adequate training.

Environmental Protection:

An important advantage of a space hangar would be its potential capability of protecting the maintenance worker from the hostile environment. There are several significant conditions which exist in a space environment from which man must be protected or must avoid, these are as follows: hi vacuum, micro-meteoroids, extreme temperatures, extreme lighting conditions, ultraviolet radiation, Van Allen type radiation. The authors believe that a maintenance hangar could provide protection for all of these except the Van Allen type radiation. A pressurized hangar would certainly add protection from the hi-vacuum of space. In fact, a man could remove his space suit and work in a shirt sleeve environment. Although a further reduction in repair time could be made by this technique, it is not recommended because of the possibility of the hangar being rapidly decompressed by puncture. The pressurized hangar would also provide protection to the worker in the event of a space suit failure during a maintenance task, for example, a tear. Therefore, it is concluded that a hangar should have the capability of being pressurized to 3-5 psia. which is the normal pressure utilized in current space suits.

As the pressurization of the hangar is required for protection from the hi-vacuum, it is apparent that the structural designer must utilize materials of adequate strength to sustain the inflation stress. The hangar will provide additional protection from micro-meteoroids, simply because of the increased mass of material it will decrease particle penetration. These particles dissipate most of their energy in a relatively short distance. The depth of penetration of a particle is determined by the following mathematical relation:

$$h = r\, d\left(\frac{P_p}{P_t}\right)\ \left(\frac{v}{c}\right)$$

where h = depth of penetration. d = diameter of the impacting particle. P_p & P_t = density of particle and targer material respectively. v = velocity of particle. c = the velocity of sound in the target material. r, , = empirical constants.

It is well known that man will encounter extreme ranges of temperatures in space; however, it is also known that a very effective technique for controlling temperature in space is by using ratios. In other words, a temperature control coating system would affectively solve this problem. Such a system could be easily applied to a maintenance hangar. Coatings for temperature control are already in operation on several satellite systems. Therefore, a definite requirement for a temperature control coating should be included in the criteria for a space maintenance hangar.

The American Lighting Society recommends that a minimum lighting of 20 foot candles be provided for most earth tasks. On earth this is easily accomplished by direct or indirect lighting; however in space man can be exposed to total darkness or to blinding sunlight. Therefore, the maintenance hangar could provide a shield from these various extremes and could also provide controlled artificial lighting. Some authorities recommend significantly higher levels of lighting. Therefore, any maintenance hangar must provide the capability of artificial lighting.

One of the more perplexing problems in space is the protection from ultraviolet radiation. A hangar if aluminized could stop the majority of all dangerous U.V. radiations. Therefore, the hangar must provide protection from ultraviolet radiation which can seriously damage man's eyesight.

The next problem is Van Allen type radiation. Although the hangar would provide some protection, it would be insignificant. Therefore, it is not recommended that a space maintenance hangar be considered for use in the Van Allen radiation belt.

A space maintenance hangar must be structurally sound and if punctured by a small particle should not fail catastropically. Such a hangar should have a high strength to weight ratio, have a small launch package volume, be reliable and easily deployed.

CONCLUSION

The authors have concluded:

1. A significant reduction in time to perform a maintenance task can be achieved through the utilization of a space maintenance hangar.

2. A maintenance hangar can offer protection from the hi-vacuum, micrometeroid, temperature extremes, lighting extremes and ultra-violet radiation existing in space.

3. The most logical solution towards launching a maintenance hangar into orbit would be an expandable hangar either of the inflatable structure or expandable honeycomb type.

4. Space maintenance is a necessary requirement in the development of a successful operational space system.

5. Space maintenance should not be conducted while in the Van Allen Belt or during high radiation solar storms.

6. Research in this area should be continued.

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EXPANDABLE SELF-RIGIDIZING

HONEYCOMB STRUCTURES

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I. INTRODUCTION

The advent of manned space exploration has brought about an increased emphasis on strength-to-weight ratios and launch package volume. The time table of exploration calls for very large diameter solar collectors, space stations, and shelters to be transported into space. The transportation of these large size structures requires a technique of either assembling small modular components in space or the development of an expandable structure. It should be pointed out that the above discussion also applies to limited war shelters and structures. Viron Division of Geophysics Corporation of America and Archer-Daniels-Midland Corporation under Contract AF 33(657)-10409 are developing a technique which will not only satisfy the requirement of small package volume, but will also provide structures of very high strength-to-weight ratios. This paper will discuss the structural concept, materials investigations, fabrication techniques, and finally, actual structural rigidization results. This initial effort has been directed towards producing terrestrial and aerospace shelters and space solar collectors.

II. STRUCTURAL CONCEPT

This concept utilizes a woven fiberglass corrugated or honeycomb type core which is impregnated with a plastic resin system. This resin system is initially flexible, but on command will rigidize into a rigid fiber-plastic composite. The exterior surface skins and interior cell walls or corrugation webs are impregnated with the subject resin system. Rigidization of the resin system can occur from radiation cross linking, plasticizer boil off, or vapor phase catalysis. The latter two have been found to be the most suitable under

this research effort. This structure when rigidized will produce results comparable to the well known excellent rigid conventional honeycomb properties. This concept was originally conceived by S. Allinikov and F. Forbes of the Research and Technology Division.

III. DESIGN GOALS

This feasibility study was undertaken to design, fabricate, and rigidize a 5 ft. diameter solar energy concentrator and a 7 ft. diameter cylindrical space shelter using expandable honeycomb type structural material. The solar collector was to weigh about 0.2 lb./ft.^{2} of reflective area and have a reflectivity of 0.86 to the solar spectrum. The reflective surface was to be within $\pm$ 1/4° of a paraboloid over 99% of the surface.

The space shelter was to be designed to withstand a roof loading of 100 lb./ft^2 and be capable of withstanding in internal pressure of 14 psi. The cylindrical portion was to be 5 ft. high with a flat bottom and a hemispherical dome.

IV. MATERIALS INVESTIGATION

A. Resins

There are several techniques for the chemical rigidization of a honeycomb structure. Some of these under evaluation are as follows: A resin system which cross-links under the influence of a vaporized catalyst which can also be used as part of the inflation media, plasticizer boil off from a solvated polymer system, and polymer cross-linking through ultraviolet, ionizing, or thermal energy. The first two approaches have been under evaluation in this investigation.

The results obtained during the course of this investigation include: Operable resin systems producing rigid expandable structures under vacuum conditions employing vapor phase catalysis which have been demonstrated. Of the several thermoset resin systems investigated for vapor phase catalysis cure, urethane and epoxy resins show most success.

A one-component system containing an isocyanate terminated urethane prepolymer which is cured by moisture (water vapor) has been developed which shows the following results:

1. Cure times of approximately 1-2 hours. This was accomplished on small scale models in a laboratory vacuum environment using water/triethyl Amine as curing agent.
2. Minimum resin saturation of approximately 20% by weight has been shown to be feasible.
3. Tensile strengths of one ply laminates prepared from 181 fiber-glass cloth of approximately 25,000 psi cured by moisture in 50% humidity for 3 days at 75°F.
4. Flexural strengths of 3 ply laminates prepared from 181 fiber glass cloth of approximately 12,000 psi cured in 50% humidity for 3 days at 75°F.
5. A low degree of tackiness after solvent loss from an impregnated structure. This factor eliminates the need of outer barrier film.

The commercial epoxy resins were rapidly screened and upon evaluation the Dow DEN 438 exhibited most desired properties (cure time, hardness, thermal stability). Optimum cure times recorded in the program under vapor catalyst (propylene diamine) conditions were 6-7 hours. This was only accomplished by employing an experimental accelerator which was premixed directly with the resin before application. Without the accelerator additive, vapor cure times were in the 14-16 hour range. Data of interest on the DEN 438 impregnated fiber glass 181 includes the following:

1. Resin saturation of 40% by weight.
2. Tensile strengths of 17,000 psi on three ply laminates cured by 20% propylene diamine vapor in a time period of 16 hours in a hard vacuum.
3. Flexural strengths of 23,000 psi on three ply laminates cured in a similar manner as outlined above.

An operable resin system producing a rigid expandable space vehicle employing the plasticizer boil off technique in a space environment has been demonstrated. This resin consisted of a multiple component system including polyacrylic resin, a styrenemaleic anhydride copolymer and a melamine resin. Time for complete rigidization depends on the vacuum produced; however rigidization can be achieved under two hours. Tensile strengths of approximately 20,000 psi have been recorded on single ply laminates prepared from glass fabric style 181.

The major emphasis of this investigation was conducted on vapor phase curing systems. A rapid screening of the physical properties observed from commercially available resin systems and convenient catalyst systems was undertaken. These included the following:

Resin System	Catalyst
a. Urethanes	Water, Amine
b. Epoxy	Amines, Lewis Acids
c. Furfural resins	Acidic catalyst
d. Phenylsilanes	Water, Amines

1. Urethane

The research effort was devoted to (a) characterization of urethane prepolymers including composition, cure time under vacuum conditions with catalyst, results of filler addition to resin, and .sual properties of moisture cured laminates. (b) Physical data compilation of urethane free films and urethane impregnated laminates and (c) catalyst study involving resin impregnated structures under space and terrestrial environment.

a. Characterization of Urethane Prepolymers

The basic compositions of the polymers evaluated consisted of a short diol, triol and a polypropylene glycol of medium molecular weight and a diisocyanate. The prepolymers were formulated at 50% non-volatile content to enable convenient methods of application. In the absence of moisture shelf lifes of 18 months have been observed. The free isocyanate content of these prepolymers range from 4 to 6% based on 50% solids. By varying the relative amounts of short chain and long-chain diols in the formulation, or by modifying the molecular weights of either the short-chain or long-chain diol,virtually any degree of hardness can be achieved.

Experimental data indicates that best results are obtained through using prepolymers with low % of reactive isocyanate groups. The addition of polymeric fillers enhances the strength and rigidity, however, the shelf stability of the system generally decreases.

b. Catalyst-Resin System Relationship

The following parameters effect the overall performance of the urethane:

1. Solvation of the urethane prepolymer
2. Resin-glass weight ratio
3. Cure rate variations
4. Methods of catalyst deployment

<u>Solvation of the Urethane Prepolymer</u>

The urethane prepolymers described in this report are isocyanate terminated polymers prepared by the reaction of diisocyanates with polyols. The ratio of diisocyanate to polyol is selected so that a soluble urethane polymer of molecular weight between 1500 and 3000 is formed which possesses terminal isocyanate groups. Upon application, such a polymer may then react with moisture to form a high molecular weight polymer coating. The vehicle described is most conveniently made at 50% non-volatile material content. The viscosity of this vehicle at room temperature (700 + 75 Brookfield, cps) is suitable for conventional methods of application. Saturation of glass weave fabrics (Raypan or Wimpfheimer aerospace structures) is easily accomplished through hand lay-up, spray gun technique or vacuum impregnation. The volatile component of these vehicles is a solvent system mixture of xylene/cellulose acetate or toluene/butyl acetate.

<u>Resin-Glass Weight ratio</u>

The weight ratio of resin to glass has a direct relationship to the ultimate strength to weight performance of the rigidized structure. In order to determine the minimum resin concentration required a series of experiments was conducted with varying weight ratio of resin to glass. Results indicated that 20% resin solids is sufficient to achieve sample rigidization. This is accomplished by diluting the solids content of the urethane to 40% NV by the addition of ethyl acetate before application to the glass. Hand lay up by applying the resin with a paint brush allowed a thin film of resin to be applied. This thin resin film formation enhances rate of cure as the depth the catalyst has to penetrate is reduced. This also allows for a more completely cured system. The most desirable resin to glass ratio found experimentally for large honeycomb glass fabric structures featured a resin content of between 30-40% solids. In this resin content range the strength to weight ratios were the highest.

Cure Rate Variations

Time variations in the cure rate were observed as a consequence of the volatile contents of the urethane prepolymers. A series of experiments were conducted of resin impregnated structures with and without prior removal of the volatile contents of the prepolymer. In both instances the structures were subsequently cured with water vapor at reduced pressures (.1 mm/Hg.) Removal of solvent before water vapor contact did enhance the rate of cure by a factor of two. Cure times of 4 hours were recorded. Complete solvent removal from the impregnated structures was accomplished by placing them in a 12" desiccator and evacuating (.1 mm/Hg) for 18 hours. Before opening the desiccator to the atmosphere nitrogen was bled in to provide an inert blanket in order to exclude atmospheric moisture as much as possible. Upon examination the structure was completely flexible and essentially tack free due to the formation of a surface coating. After compression of the structure it was placed in a space chamber and connected by means of tubing to a water supply. Evacuation of the system (at .1 mm/Hg) caused volatization of the water with the result that the structure fully expanded and rigidified in the 4 hour time period. A most noteworthy improvement in cure time was observed when a 90-10% weight ratio of water to triethyl amine curing system was employed. This curing system produced approximately one hour cure times for resin impregnated glass structures at room temperature conditions. Further investigation of this system with and without prior removal of solvent from resin impregnated structures indicated solvent removal is not a critical parameter. The triethylamine greatly activates the isocyanate functional group by complex formation and renders the isocyanate more susceptible to attack by the water molecule. In essence, what the addition of triethylamine accomplishes is to speed up the reaction of the isocyanate with water. The unstable carbamic acid loses carbon dioxide and forms the primary amine group which reacts rapidly with another isocyanate to form the crosslinked disubstituted urea.

Methods of Catalyst Deployment

Several mechanical devices employed to selfcontain catalyst within the resin system and to release it upon demand in the environment desired have been explored. One of these involves the use of polyethylene tubules which can be inserted between the flutes of three-dimensional integrally woven fluted panel core material (Raypan) and subsequently filled with catalyst. The ends, when sealed, give a catalyst container within the system. Such a system has shown limited operational success.

A second method that shows initial feasibility involves coating solid hydrates with a thin resin layer. Upon desiccation these materials give up to 33% of their chemically held water after three hours at .1 mm/Hg of pressure.

A third method involves coating a solid light weight absorbing material, such as vermiculite with a thin resin coating. This substance is capable of absorbing up to 80% of its weight with water. Under vacuum environment as much as 50% of the retained water can be extracted. These systems of water containment are highly efficient and add little weight to the system. The materials are coated with a thin coating of Admiral 351 which is a cyclopentadiene copolymer resin.

2. Epoxy

The vapor catalyzed curing of an epoxy system was found to be operationally feasible. Our initial effort was to screen the commercially available resins and from this data choose the most likely systems to further evaluate. This data was collected on samples cured at room temperature by direct mixing of catalyst and resin. It was felt that the most operable system under these circumstances would be most operable under conditions employing vapor catalysis. In this manner we were able to accomplish an evaluation of numerous resin-catalyst-accelerator combinations and to record optimum concentrations of each component. The system showing the most promise was the Dow DEN 438 diluted with xylene. The viscosity of DEN 438 prohibits working with it as it is commercially available. Xylene was chosen as the diluent for the following reasons: (a) Its vapor pressure (760 mm at 136-144°C) is close to that of propylene diamine (760 mm at 135.5°C); (b) xylene being a non-reactive diluent should not appreciably lower the crosslinking potential of the system as do reactive diluents; (c) the cure seems to be greater with xylene than when a resin of low viscosity (Epon 812 or DER-332) is used as diluent possibly because the curing agent is not able to penetrate before a surface cure is obtained.

DEN 438 (80 parts by weight)-xylene (20 parts by weight), the most operable system was applied to small prototypes of a space shelter. These shelters were fabricated from Raypan fluted panel core No. 403. All seams were sewn, the structure was impregnated with an equal weight of resin and propylene diamine was allowed to pass through at a pressure of 20 ± 10 microns at room temperature. After 24 hours at this pressure the shelter was removed and found to be rigid enough to stand on (a similar treatment using DEN 438 (80 parts)-Acetone (20 parts) gave only a surface cure). An incomplete cure was obtained after only 15 hours.

3. Polyacrylic

Initial investigation was conducted on an experimental ADM polyacrylic resin 5803-20-6, containing 60.3% solids by weight diluted in a 75/25 xylene-butanol solution. This product impregnated

than 10 oz./yd^2 resulted. A quasi-honeycomb can be woven in this manner by placing the rows of pile perpendicular to each other. A much heavier material can be obtained by using, for example, fiberglass yarn and placing the interwoven drop threads at nearly any given density.

V. EXPERIMENTAL WORK

A. Six Inch Cubes

In the early stages of the program a series of 6 inch cubes were vacuum cured by vapor catalysis. The structural material was fiberglass woven into a three dimensional fabric with a fluted core. The cubes were impregnated with an epoxy and a beaker containing an aliphatic amine was placed inside the structure. Cure times of 16 hours were obtained with a relatively low molecular weight epoxy and a large excess of amine.

Higher molecular weight epoxies gave longer cure times but it was found that large excesses of amine could be used to speed up the rate of cure. Polyethylene liners were used in some of the experiments to slow down the escape of amine vapor and it was noted that the rate of cure was not appreciably affected.

A three dimensional Nylon material with drop threads between 2 skins was also used to fabricate cubes. Very good inflation was achieved but strengths of the cured structures were low compared to the fiberglass cubes.

B. Model Shelters

Numerous 8.4 inch diameter model shelters were vacuum cured at 0.1 mm Hg with epoxies and urethane resins impregnated onto both the nylon and fiberglass structural material. Figure 1. Cure times as low as 7½ hours were obtained with the use of a special accelerator developed by ADM and a low molecular weight epoxy with a viscosity of 50-90 centipoises.

After the development of a fast curing urethane resin by ADM, it was decided to devote all further work during this feasibility study to optimize its use. A combination of the urethane and three dimensional fiberglass fabricated into 6 inch cubes resulted in 2 hour cures. Water and an amine placed in a beaker inside the cube served as both inflation and curing agents.

C. 3½ Foot Shelters

Several 3½ foot shelters were expanded and rigidized in a 5 foot diameter vacuum chamber. Figure 3. The structural material was the fluted core fiberglass which had been impregnated with the urethane resin. It was found that a polyethylene inner liner was necessary for proper inflation because of the porosity of the structural material. The water and amine were piped into the house from a three liter flask. Figure 2. Cure times ranged from 4 to 6 hours. A 3½ foot shelter was inflated and cured in the atmosphere by introducing air and steam to the interior. The fabric was vacuum impregnated with a urethane resin and placed in a special vapor barrier container and bathed in an inert nitrogen atmosphere to extend shelf life. Figure 4. The unit was fabricated with an eliptical door which had been previously rigidized [illegible] a 2 inch section around the door opening to serve as a framework. A neoprene gasket served as a seal. Cure time was about 6 hours.

on glass style 181 or Fortisan NW 372 and subsequently desiccated to 10^{-3} mm Hg rigidified rapidly by plasticizer release. The rigidified samples were coated with an extremely brittle, blistered film which offered little strength. In addition the adhesion of the polymer substance to both substrates was poor. The prepared samples were inadequate for physical testing.

The polyacrylic resin modified by the addition of styrene-maleic anhydride copolymer (3000 M-wt) and a melamine resin gave a plasticizer release system which exhibited the most promising performance. The advantages of this system include (1) relatively better adhesion to glass compared to the polyacrylic resin unmodified. (2) reduction in blistering and forthing and (3) a more rigid surface. The adhesion of resin to glass was improved by synthesizing an acrylic resin high in free carboxyl. The increased polarity attributable to the carboxylic acid group increases surface adhesion of the resin to a substrate such as glass. The rigidity afforded by removal of the plasticizer from this system has been greatly improved by the addition of 20% by weight of styrene-maleic anhydride copolymer and 5% by weight melamine-formaldehyde resin. For ease of impregnation this mixture was solvalized with acetone and applied from solution to give a highly uniform deposit of resin and thus a thin surface film. This had the overall effect of decreasing the tendency of the resin to blister during desiccation. Cure times employing vacuum conditions are under two hours for maximum rigidity.

B. Structural Material

An extensive search of the weaving industry revealed some unique three dimensional expandable structural material for aerospace use. It also brought out the fact that at the present time no one in the United States manufactures a completely flexible true honeycomb core fabric. A true honeycomb sandwich material consists of an orderly array of cells oriented perpendicular to the skins and the optimum honeycomb is one in which the cells are woven integrally with the same threads as the skins. If a fiber or yarn such as fiberglass were used to weave this fabric, a flexible material would result which could then be impregnated with a s[illegible]able resin, folded and packaged, and at a later time inflated and ri[illegible]idized. Although we have found the capability to produce such a material the machine or loom will not be in production for about 24 months.

Several fabrics which closely resemble a true honeycomb weave were utilized in the current feasibility study. A flexible three dimensional fiberglass material characterized by flutes integrally woven between skins was used with a resin in the construction of space shelters. The flute walls ran along the length of the fabric and were placed parallel to each other such that the flute cross-section was an isoceles triangle.

A member of the velvet industry provided certain three dimensional fabrics well suited for expandable self rigidizing structures. This material had two skins connected by drop threads or pile. The pi[illegible] could be a variety of patterns, densities, and lengths. For example, one fab[illegible] used to construct solar concentrators was woven from Nylon with two skins separated by rows of drop threads 1/2 inch apart. A fabric weighing less

than 10 oz./yd^2 resulted. A quasi-honeycomb can be woven in this manner by placing the rows of pile perpendicular to each other. A much heavier material can be obtained by using, for example, fiberglass yarn and placing the interwoven drop threads at nearly any given density.

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FIGURE 1. VACUUM CURED 8.4" MODEL SHELTER

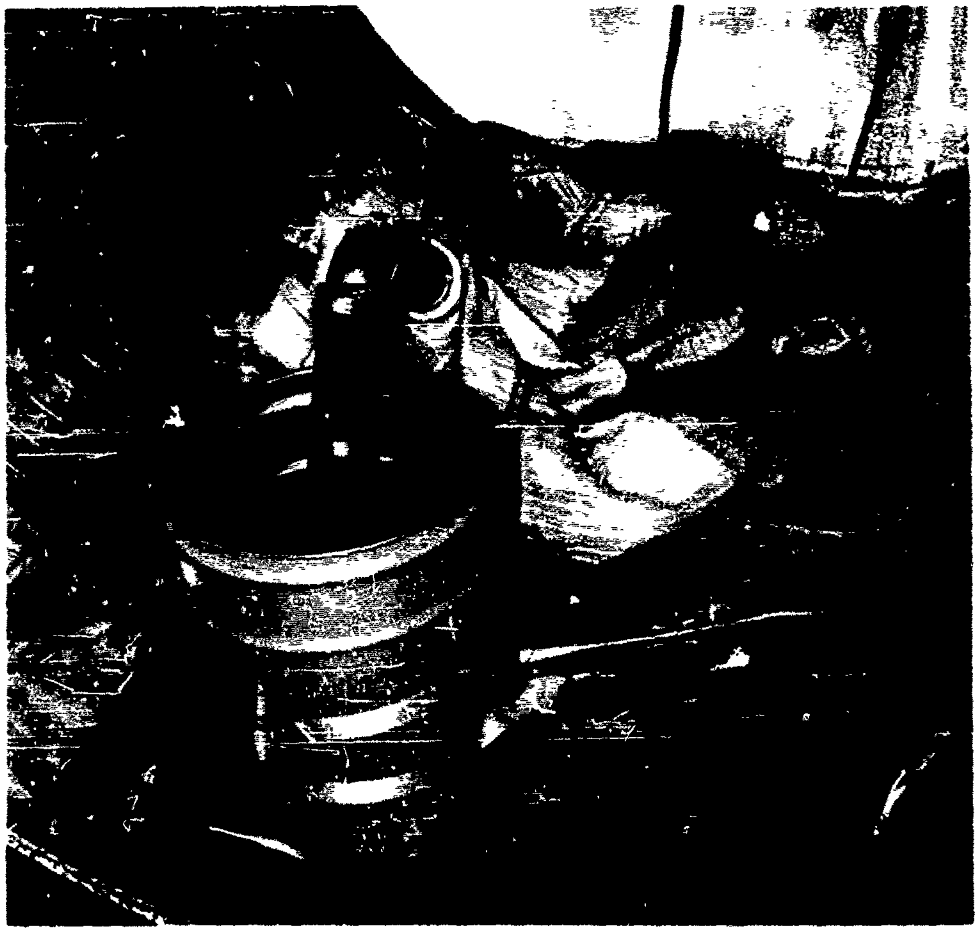

FIGURE 2. RESIN IMPREGNATED 3½ FOOT SHELTER READY FOR INFLATION

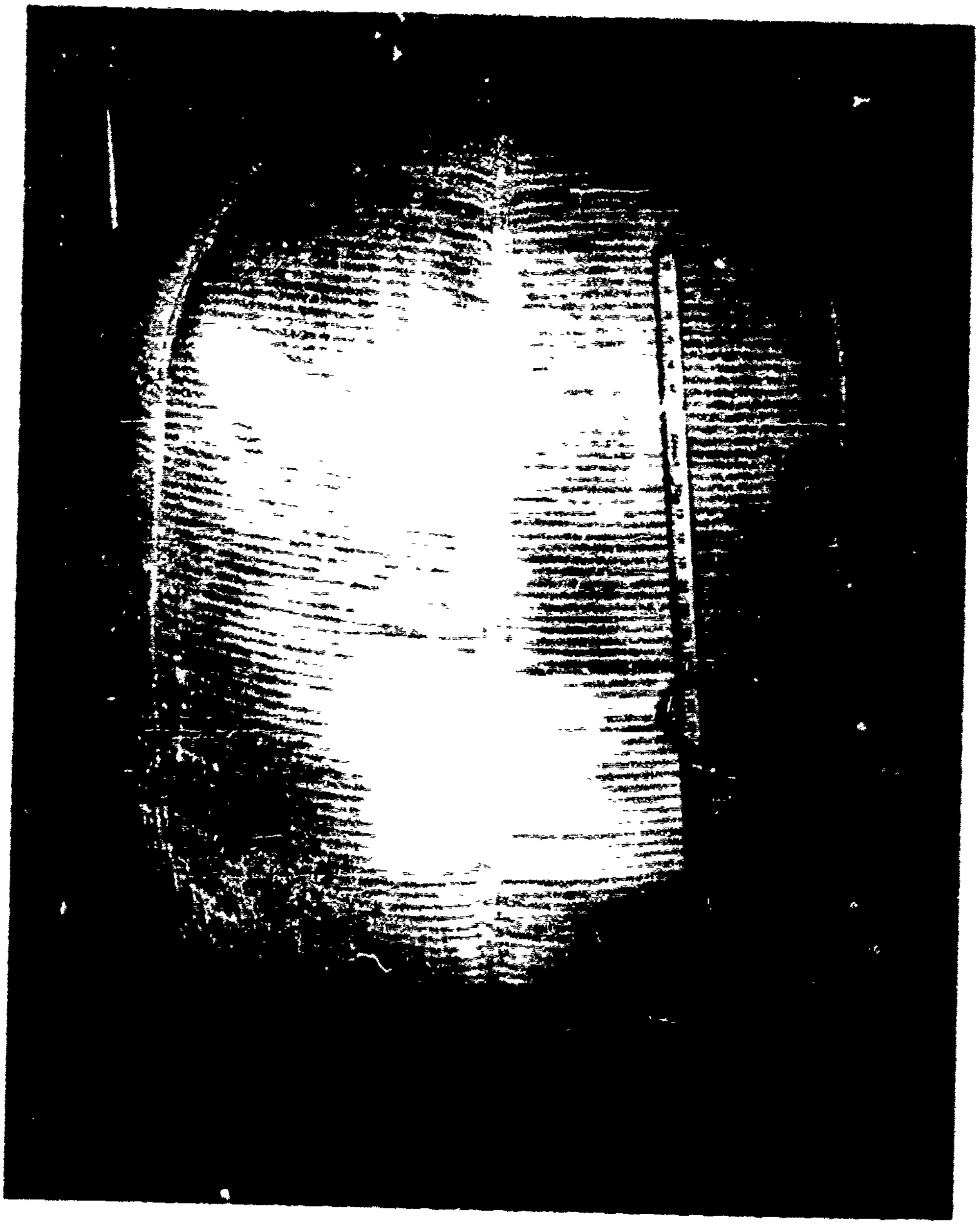

FIGURE 3. 3½ FOOT VACUUM CURED MODEL SHELTER

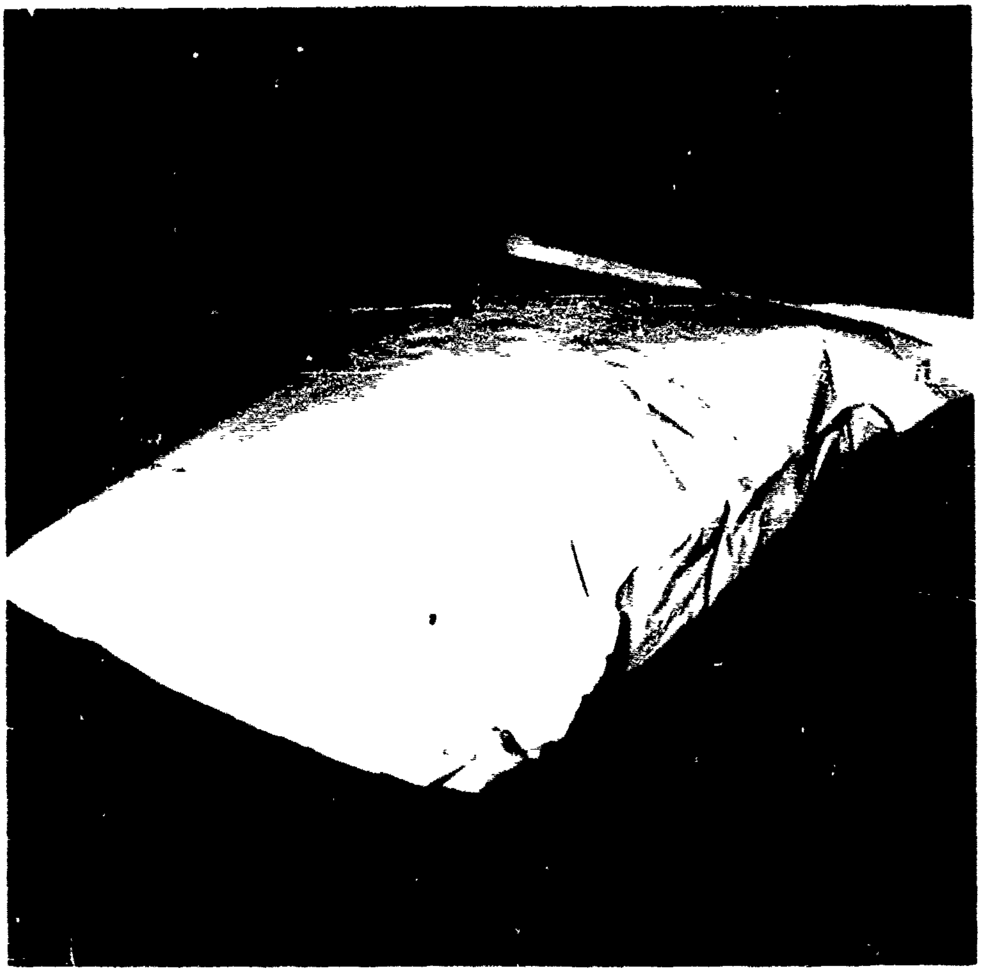

FIGURE 4. PACKAGED 3½ FOOT SHELTER

D. 7 Foot Shelters

Seven foot space shelters were fabricated and cured in a Wright Field vacuum facility. The structural material was the same as that used in the 3½ foot diameter models. The shelter was made from 8 shaped gores sewn together by means of a lap and double strap seam. Total weight of the impregnated structure was about 100 lbs. Packaged volume was 2 ft.^{3}, inflated volume 250 ft.^{3}. Inflation and subsequent cure with water vapor in 8 hours yielded a rigid structure. Figures 5 through 7 show the packaged, impregnated, and cured stages.

E. Solar Collectors

In order to establish a procedure, the early work consisted of bench curing 8 inch diameter structures. A preformed aluminized Mylar film was first coated with a layer of rigid resin and 181 glass cloth. The rigidized composite had severe show-through from the weave pattern of the cloth and also had warped. A three dimensional nylon with dense pile eliminated the warp and a thicker aluminized Mylar film eliminated much of the show through. Attempts were than made to preform the nylon into the desired shape with good success. Adhesion of rigid resins to the Mylar was poor but this was overcome by applying a very thin coat of a polyvinyl acetate emulsion to the film.

Excellent reflective surfaces were obtained by bench curing 8" concentrators with rigid resin as a cushion for the three dimensional structural material. After finding a method of preforming the aluminized Mylar and the nylon fabric, the rigid resins were replaced by flexible resins to serve as a cushion and also as a bonding medium.

Vacuum cures were then attempted by impregnating the nylon with an ADM polyacrylic resin and removing the solvent at a reduced pressure. Good cures were obtained in two hours and the resulting structure had sufficient rigidity to hold its shape.

Several 24 inch diameter models were vacuum cured in the same manner. Three dimensional nylon with a dense pile between skins 5/8" apart was used in this series of tests.

When placed in direct sunlight the reflective surface produced an image of about one inch diameter which would instantly ignite paper or cloth. This type of structure however, had the drawback of too much weight due to the nylon fabric as well as the wasted solvent system necessary for rigidization.

After the development of the polyurethane resin system together with the design of a new expandable honeycomb type material, it was possible to build much lighter units. The structural material was woven from nylon with a total weight of about 10 oz./yd.^{2} Parallel rows of drop threads were placed 1/2 inch apart between two skins. The steps involved in rigidizing these concentrators were as follows:

1. An aluminized Mylar film was preformed to the desired shape.
2. One to three coats of a flexible or semi-flexible resin was brushed over the Mylar surface and allowed to cure. It was found that a completely flexible resin had good adhesion to the Mylar and the adhesion was still satisfactory when the flexible resin was blended with small amounts of a rigid resin. This made it possible to eliminate the thin coat of polyvinyl acetate.

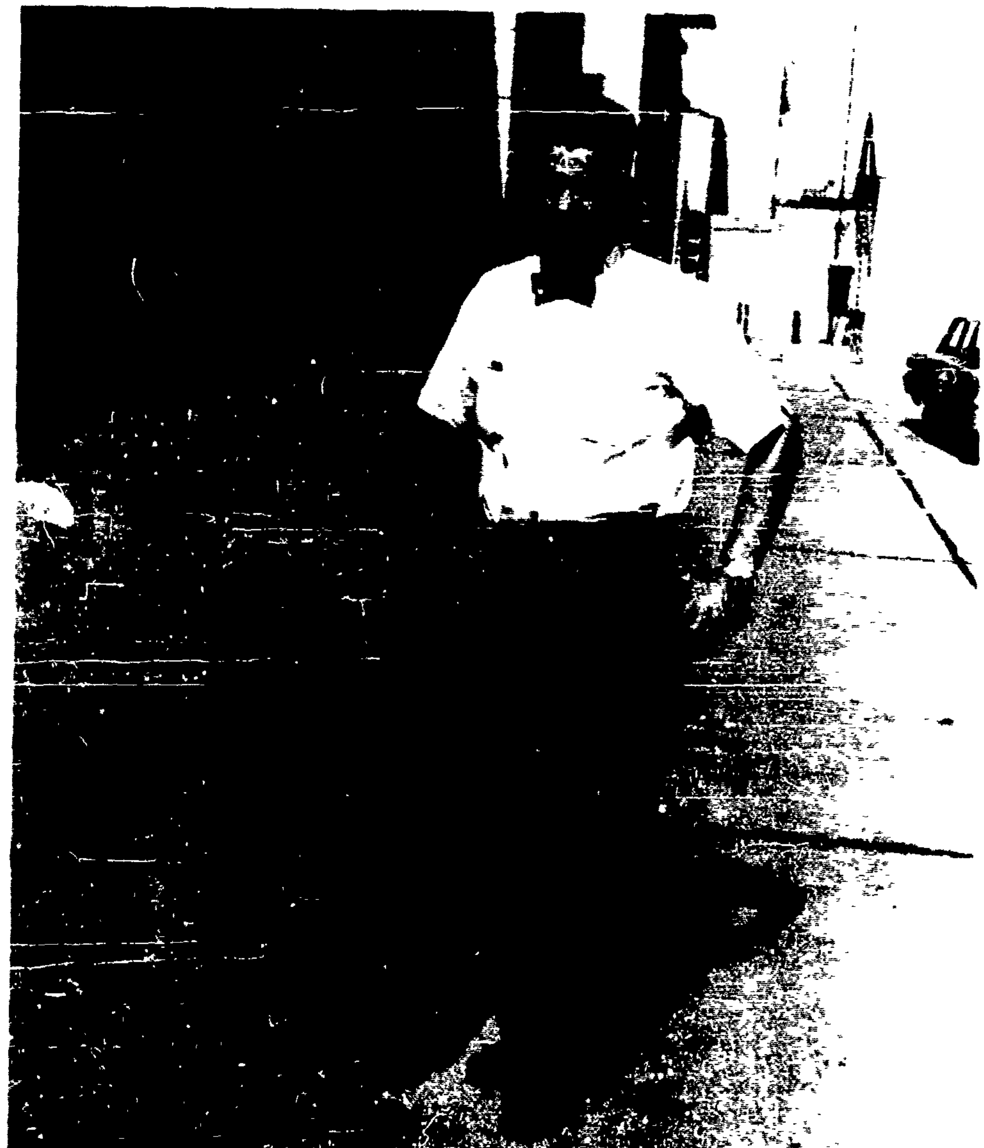

FIGURE 5. PACKAGED 7 FOOT STRUCTURE

FIGURE 6. IMPREGNATED 7 FOOT STRUCTURE

FIGURE 7. CURED 7 FOOT STRUCTURE

3. Another coat of flexible resin was applied and when a desired tackiness was achieved, the preformed nylon fabric was laid over it. This coat provided a good bond between the nylon and the cushion layer.
4. The nylon structural material faces had been sewn together near the outer edge to permit inflation of the core. Valves were attached to the outer skin for entrance of the inflation and rigidifying vapors.
5. The nylon structural material was then impregnated with the urethane resin.
6. Water and an amine were placed in flasks which were connected to the valves on the structural backing material.
7. The Mylar film was held in place by a set of rings. A cover film was placed under the reflective surface which provided a means of pressurizing the structure to the desired shape.
8. The assembly was placed in a vacuum chamber for two hours. Figure 8. Pressure inside the cover film was regulated by means of tubing from outside of the chamber. Another tube led from inside the cover film to a manometer so the internal pressure could be monitored and regulated. Good cures were obtained in the 2 hour period. Figures 9 and 10. The greatest problem encountered was deformation of the reflective surface due to wrinkling of the structural material and show through of the core walls.

Two five foot diameter solar energy concentrators were similarly cured with a urethane resin at Wright Field. The structures were sufficiently flexible for packaging, cured in two hours, and had a fair reflective surface. Figures 11 through 14 show the collector in the packaged, impregnated, and cured stages.

VI. CONCLUSIONS

A. The feasibility of utilizing expandable self rigidizing honeycomb structures for both space and terrestrial application has been successfully demonstrated.

B. In addition the feasibility of using this same technique for the fabrication of solar energy concentrators with their extreme requirement for reflective surface regularity has also been demonstrated.

VII. RECOMMENDATIONS

Continued effort is needed toward optimization of structural design and rigidization techniques.

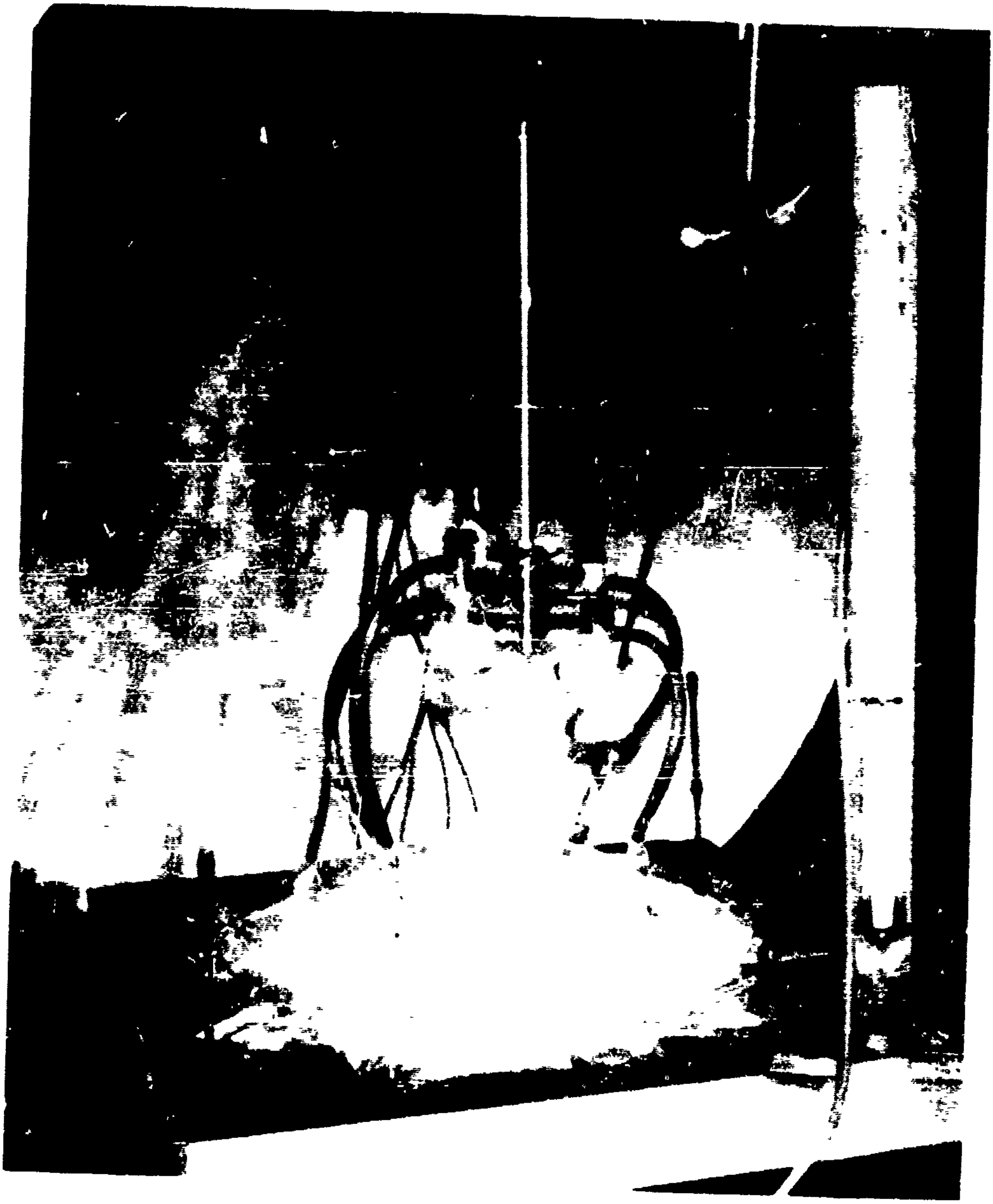

FIGURE 8. 24" SOLAR COLLECTOR READY FOR VACUUM CURE

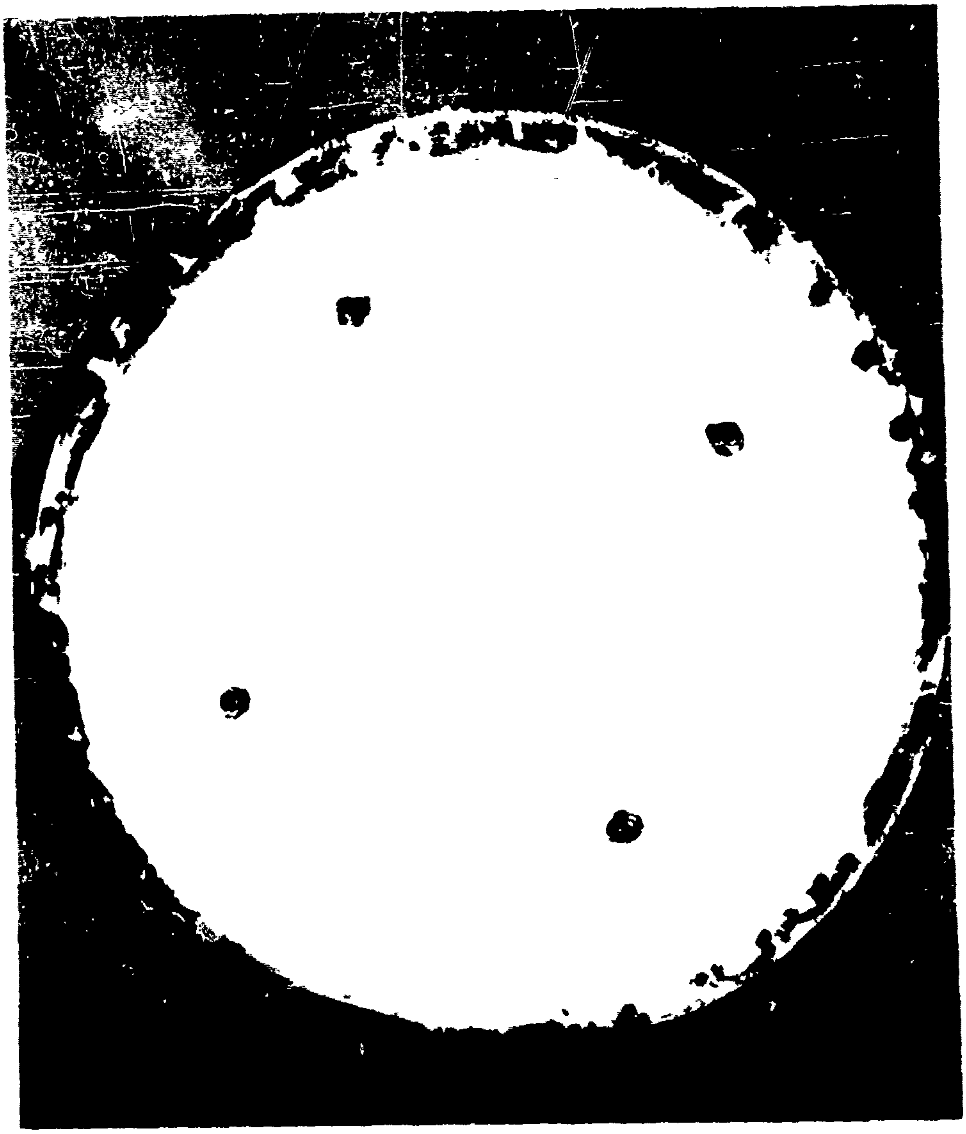

FIGURE 9. 24" SOLAR COLLECTOR SHOWING VACUUM CURED STRUCTURAL MATERIAL

FIGURE 10. REFLECTIVE SURFACE OF A 24" SOLAR COLLECTOR

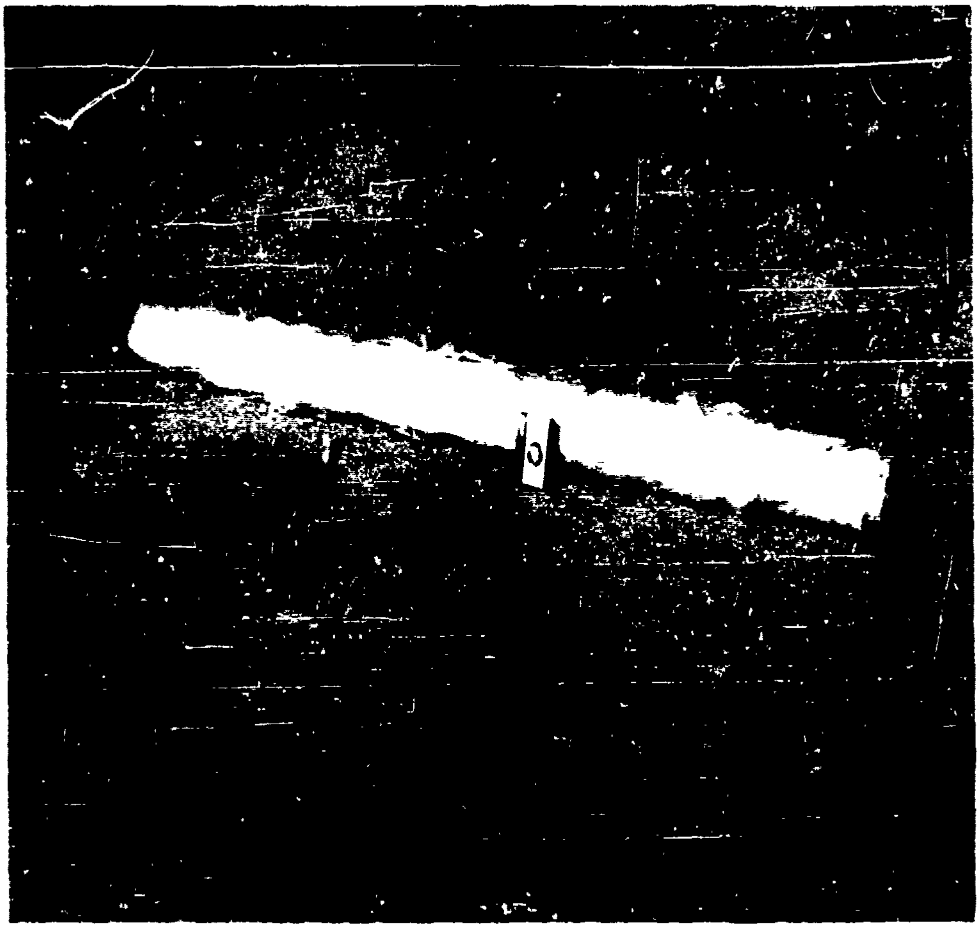

FIGURE 11. PACKAGED 5 FOOT SOLAR COLLECTOR

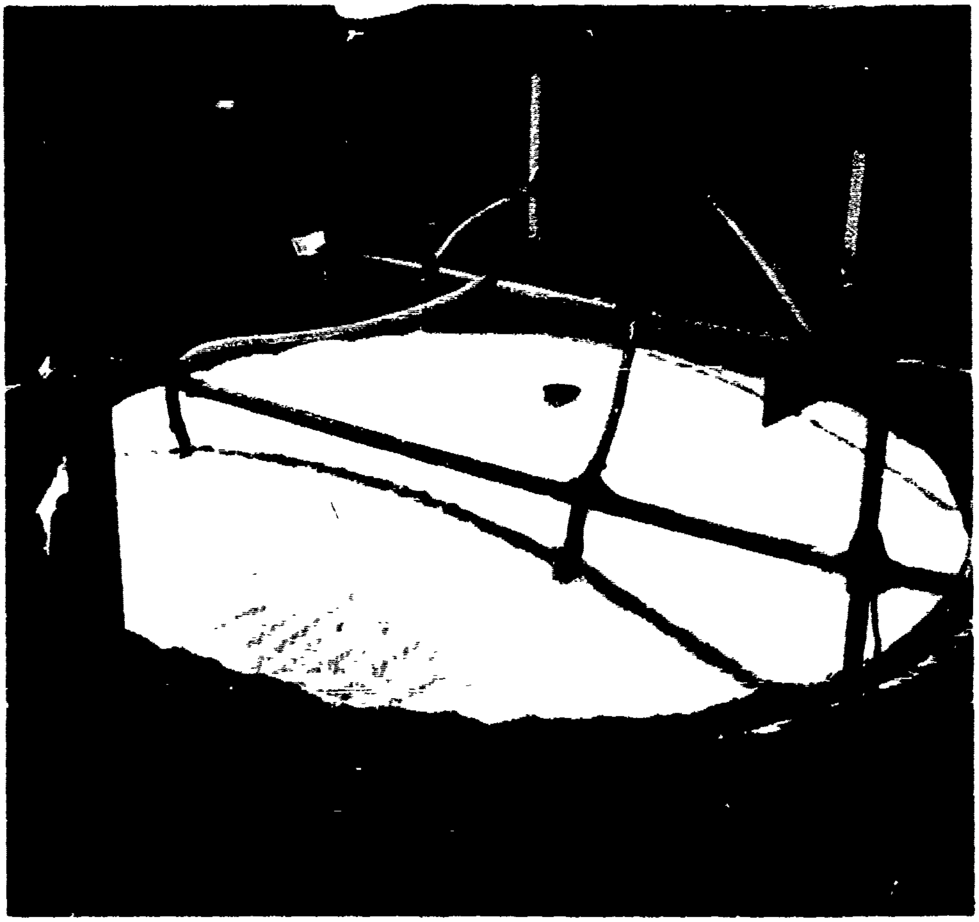

FIGURE 12. IMPREGNATED 5 FOOT SOLAR COLLECTOR

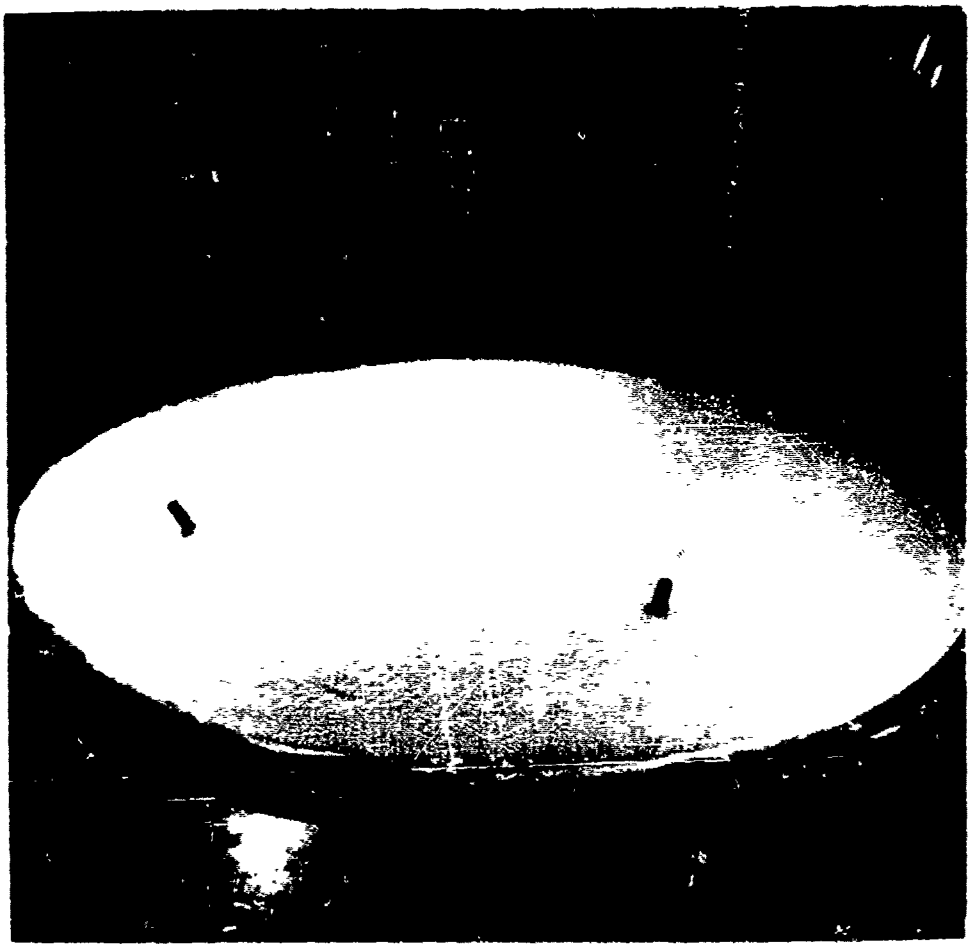

FIGURE 13. CURED 5 FOOT SOLAR COLLECTOR

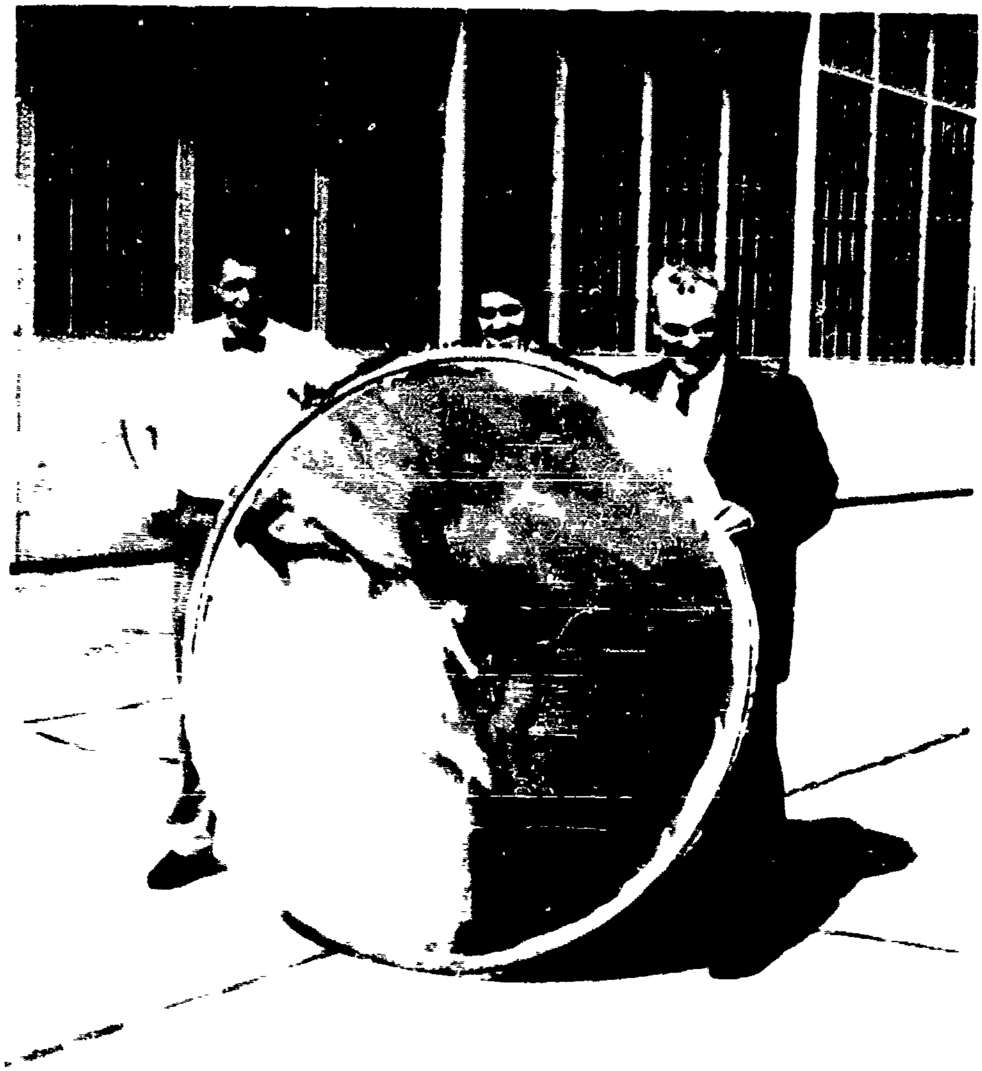

FIGURE 14. CURED 5 FOOT SOLAR COLLECTOR

GELATIN, A RIGIDIZING MATERIAL FOR AEROSPACE STRUCTURES

by F. W. Forbes, Air Force Aero Propulsion Laboratory
S. Allinikov, Air Force Materials Laboratory
C. Koons, Experimental Fabrication Division

I. INTRODUCTION

General: The National Aerospace Program has brought about the requirement of placing large size structures into space, such as passive communication satellites, solar collectors, space stations, and extraterrestrial shelters; however, this requirement poses a tremendous problem when considering the problems of boosting into space such large size structures. Consequently, materials and structural research engineers have conducted research in expandable structure concepts, such as foamed-in-place, gas catalyzed systems, plasticizer boil-off, and radiation cross-linked structures. The authors became interested in gelatin as a possible rigidizing material after preliminary in-house investigations showed gelatin to exhibit potentially high tensile and modulus of elasticity values, as well as, a high resistance to ultra violet radiation. Immediately two programs were initiated. Materials Laboratory initiated a contractual materials property investigation with Monsanto Research Corporation, Dayton, Ohio, under Contract No. AF 33(616)-8283, and the Aero Propulsion Laboratory initiated an in-house program which has demonstrated the feasibility of rigidizing space structures with gelatin.

Purpose: The purpose of this paper is to disseminate the significant research of both Air Force Laboratories' programs, and to stimulate other materials and structural investigators to consider gelatin for either future research or applications.

II. MATERIALS INVESTIGATION

Selection of Gelatin for Research Program: As previously discussed, many different types of materials have been investigated to date; however, very little research, if any, has been conducted on water soluble natural plastic resin systems. There are a number of such resin systems, for example, the proteins. Gelatin, a protein, was selected for an in-house materials study on the basis of its commercial availability, and highly polar nature. Mr. Allinikov prepared gelatin films and exposed these films to ultra violet radiation, which did not visibly affect them. Mr. Allinikov and Mr. Forbes, then both of Materials Laboratory, prepared several films of gelatin and conducted preliminary tensile tests on same. A tensile strength of 13,500 psi was obtained; however, later research has proven this figure to be considerably low. Based on these results it was felt that a research program on gelatins properties should be undertaken; in addition, it was noticed that when gelatin was plasticized

by water or other plasticizers it was foldable. Thus, it was also considered that a program should be initiated on possible expandable structure applications of gelatin. Research was initiated to determine, tensile strengths, modulus of elasticity, ultra violet radiation effects, electron radiation effects, effects of plasticizer, effects of cross-linking on water solubility, gelatin fiberglass laminate characteristics, fabrication techniques for expandable structures, and finally packability.

Composition of Gelatin: Most likely scientists ate gelatin long before their scientific curiosity prompted research in this area. A great deal of the colloid chemistry concerning gelatin is known. Recently developed analytical techniques have even expanded this information. The amino acid building blocks of the gelatin, are known, Table I summarizes these amino acids.

Table I. Summary of Amino Acid Building Blocks

(Residues of amino acids per 1000 total residues)
Source: Eastoe, J. E., Biochem. J. 61, 589 (1955)

Amino Acid	Type of Amino Acid	Residues
Alanine	Neutral α amino acids with R = aliphatic	110.8
Glycine		326
Valine		21.9
Leucine		23.7
iso-Leucine		9.6
Phenylalanine	Neutral α amino acid w/R containing unsubstituted aromatic systems	14.4
Tyrosine	Neutral α amino acids with R containing hydroxyl groups	3.2
Serine		36.5
Threonine		17.1
Methionine	Neutral α amino acid with R containing sulfur	5.4
Proline	Neutral α amino acid with R cyclized: Imino acids	130.4
Hydroxyproline		95.5*
Arginine	Basic α amino acids	48.2
Histidine		6.0
Lysine		26.2
Hydroxylysine		5.9*
Aspartic acid	Acidic α amino acids	46.8
Glutamic acid		72.0
Amide Groups		40.8
Total N %		18.3
Mean Residue Wt.		91.3

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*Contains 1 hydroxyl per molecule.

Figure I, and 2, shows the effect of water concentration in the gelatin film on the modulus of elasticity of various temperatures. It appears that a water concentration of about 4% gives optimum strength results at elevated temperatures.

The detailed or exact amino acid sequence of the molecule of gelatin is not known; however, limited sequence and limitation on permissible sequences are known:

Gly. Pro. Hypro. Gly. appears to be a frequent and important sequence.

The sequence - Pro. x Gly. Pro. - is deemed necessary for susceptibility to enzyme action. Hydroxyproline may be substituted for proline in this sequence.

A large number of the peptide fragments in gelatin contain 30-38% glycine (1 in 3 residues).

A few peptides contain greater than 38% glycine and others contain no glycine over a span of 4 residues.

Proline - and hydroxyproline - rich fragments are deficient in diamino and dicarboxylic amino acid residues. Fragments containing a large number of diamino and dicarboxylic amino acid residues are generally poor in proline and hydroxyproline.

A collagen-to-gelatin transformation may be accomplished by gentle heating or by the action of urea, sodium thiocyanate or lithuim bromide. These mild treatments produce gelatins of uniform molecular size. Commercial gelatins, produced by the action of acids or bases have a wider molecular variation and are somewhat more degraded; however, commercial gelatin was utilized in this program because of its availability. Commercial gelatin costs from 35 cents to 75 cents per pound, depending on the purity. The latter priced gelatin is utilized in various food applications.

The structure of a collagen has been described as "a triple chain coiled coil structure stabilized by one or two interchain peptide hydrogen bonds per three residue repeating units." (1) Gelatin molecules may be considered to be fragments of this type of structure with some loss in interchain binding.

Structural Properties: The Monsanto Research Corporation under the sponsorship of the Air Force Materials Laboratory determined the tensile strengths of various acid and base processed gelatin films. As mentioned previously, gelatin films prepared early by the authors exhibited strength as high as 13,500 psi; however, because of trapped air bubbles and edge effects, this figure was proven pessimistic.

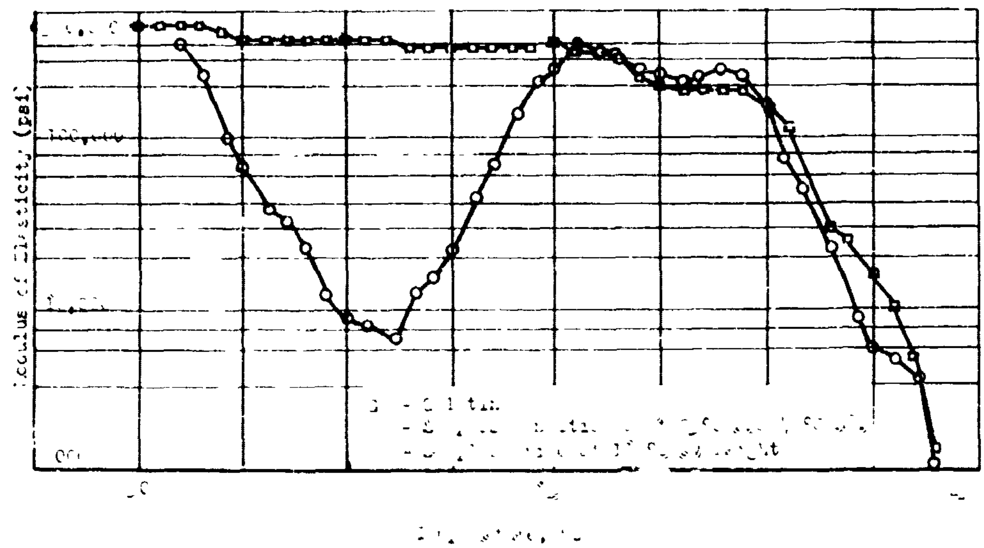


Figure 1, Effect of Water Content of Gelatin on Modulus of Elasticity at Various Temperatures

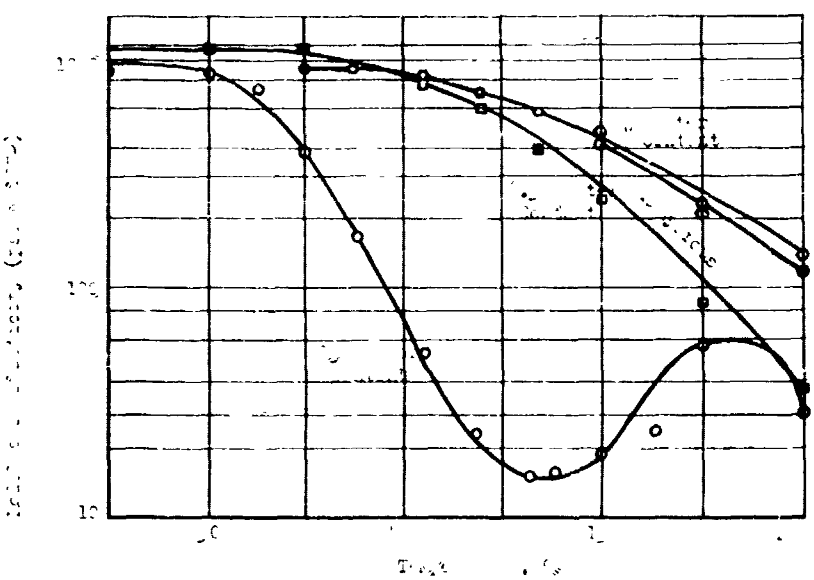


Figure 2. Modulus of Elasticity vs. Temperature

Gelatin cast films are usually produced by adding gelatin to one half the required water at about 18° C. The gelatin is soaked in the cold water for about 15 minutes, then the remainder of the water at 60° C., is added. This solution is held at this temperature, 60° C., and is gently agitated until the gelatin particles have gone into solution. A very small agitator must be utilized in order to prevent entrapment of air bubbles in the solution. Satisfactory films have been cast with solid contents ranging from 1 to 40 per cent of the gelatin water solution. Films can be easily cast and later removed from polished metal plates. Gelatin films have the same tendency to shrink and warp during curing as do other cast plastic films. The solution to this warping lies in properly balancing the ratio evaporation to the rate of migration of water to the gelatin surface. For example, a 20% solids particle surface film cast to a thickness of .080", cured under a temperature of 23 - 25° C., and kept at relative humidity of 40 will not warp. Table 2, shows typical film thickness shrinkage properties of gelatin.

Table 2, Gelatin Film Thickness Shrinkage

Solids	Cast Film Thickness	Cure Time	Cured Film Thickness
20	.080"	18 hrs	.010" to .015"
40	.162"	42 hrs	.040"

There are a number of non-aqueous solvents for gelatin. These are as follows: N - Methylformamide, Formamide, Acetic Acid, Formic Acid, Phosphoric Acid, O - Chlorophenol, p - Chlorophenol, 2 or 4 or 6 - Trichlorophenol, p - Fluorophenol, Aqueous, N - Methylacetamide, and Aqueous Methanol (33% H_2O).

ASTM tensile coupons are cut from gelatin cast films after conditioning at relative humidities ranging from 40% to 100%. This minimizes the majority of edge crazing caused from cutting. Tensile strengths as high as 21,000 psi have been obtained from these cast gelatin films. Table 3, shows a comparison of properties of cast cured gelatin films at 25° F., to other well-known organic films.

Table 3, Comparison of Gelatin, Mylar, H Film at 25° F.

Tensile Strength (psi)

Gelatin	Mylar	H Film
21,000	23,000	20,000

The mechanical properties of plasticized gelatin films are shown in Table 5. Glycerol was used as the plasticizer in these tests. The elongation of a gelatin film virtually unaffected by water concentrations up to 24% by weight.

Table 4. Gelatin Film Water Content Effects On Elongation

% Water	% Elongation at Failure
0	4
13.5	5
24	6
34	23
71	36

Table 5. Plasticizer Effect On Film Strength

% Glycerol	Yield(psi)	% Elongation	Ultimate	% Elongation at Failure
15	13,030	6	13,663	20
30	4,257	6	5,013	44

Modulus of Elasticity: Gelatin films modulus of elasticity properties have been determined as a function of film thickness vs. temperature. A summary of these results can be seen in Table 6.

Table 6

Modulus of Elasticity as a Function
of Temperature and Sample Thickness

Temperature °C	Modulus of Elasticity (psi) 0.048" Sample	0.020" Sample	0.010" Sample
-75		7,500,000	
-30		1,700,000	1,300,000
-20		920,000	1,200,000
-10		850,000	1,100,000
0		850,000	1,000,000
10		780,000	920,000
20		840,000	840,000
30		920,000	780,000
40		840,000	680,000
50		640,000	580,000
60	230,000	330,000	500,000
70	150,000	170,000	500,000
80	75,000	160,000	Sample broke
90	27,000	130,000	
100	5,600	125,000	
110	~2,000	60,000	
120	~2,000	-	
130	11,000	-	
140	105,000	-	

The recovery in modulus at the elevated temperatures is attributed to initiation of cross-linking at those points. The performance of fully cross-linked films has not yet been established. An improvement of properties would be expected, if degradation has not occurred during cross-linking.

Effect of Ultra Violet Radiation: Ultra violet absorption spectra run on gelatin films indicate that gelatin is virtually transparent at wave length no greater than 250 mu; however, below this wave length gelatin absorbs ultra violet radiation. Tensile tests were conducted on gelatin samples and mylar after a 200 hour exposure to a Hanovia ultra violet lamp's radiation. It should be noted that the tensile properties reported for the gelatin are considerably lower than the values previously reported; however, these films were some of the earlier films prepared, and the film making technology had not been perfected. Table 7, summarizes the results of these tests. These results indicate that ultra violet radiation more adversely effects MYLAR than gelatin. These tests were conducted under atmospheric conditions.

Table 7, Tensile Properties of Gelatin Films and Mylar Films After a 200 Hour V.V. Radiation Exposure

	Tensile Strength		
	Before	and	After Exposure
Mylar	33,200		27,400
Acid Processed Gelatin	15,100		15,400
Base Processed Gelatin	14,900		13,700

Effect of van de Graaff Radiation: Gelatin and Mylar films were irradiated with different dosages of van de Graaff radiation in both the atmosphere and in a vacuum. Gelatin retains its strength quite well in the air at low and intermediate dosages of electrons; however, at higher dosages there is a significant reduction of gelatins tensile strength. High dose rates also make gelatin brittle and sensitive to mechanical shock (drilling holes), and sudden applications of loads. Table 8, and 9, summarize the results of these tests in air and in a vacuum.

Table 8

Effect of van de Graaff Irradiation (Electron) on Gelatin (In Air)

Material	Total Dose (rads)	Tensile Strength (psi)	% Elongation
Mylar	0	35,200	75
	5×10^6	33,800	75
	10^7	31,800	64
	5×10^7	27,700	41
Acid Processed Gelatin	0	15,100	5.5
	5×10^6	15,100	5
	10^7	15,600	5
	5×10^7	7,900	2.5
Base Processed Gelatin	0	14,900	5
	5×10^6	14,300	5
	10^7	14,500	5
	5×10^7	7,300	2.7

Table 9

Effect of van de Graaff Irradiation (Electron) on Gelatin (In Vacuum)

Material	Total Dose (rads)	Tensile Strength (psi)	% Elongation
Mylar	0	33,200	75
	5×10^6	32,500	83
	10^7	31,100	82
	5×10^7	29,600	79
Acid Processed Gelatin	0	15,100	5.5
	5×10^6	15,600	5
	10^7	15,500	5
	5×10^7	11,700	3
Base Processed Gelatin	0	14,900	5
	5×10^6	14,300	4
	10^7	13,100	4
	5×10^7	8,100	2

Permeability of Gelatin Films: Permeability of tests have been conducted by two companies. Monsanto Research Corporation reports the following results shown in Table 10. Independent permeability measurements of protein films to oxygen and hydrogen carried out by Swift and Co. research personnel, have also yielded encouraging preliminary results. Their findings to date indicate these films have one-fifth to one-eighth the transmission of hydrogen gas as compared to a "Mylar" film. Gas transmission values obtained with oxygen gas are equal to or superior than polyvinylidene chloride film, and one-fifteenth that of "Mylar" film. It is important to note that these results were obtained on films containing glycerol. Plasticized films normally become more permeable than the unplasticized form.

Table 10. Permeability Measurements

Gas	Source	Permeability $cm^3/sec/cm^2/cm\ Hg/mm \times 10^{12}$ Mylar	Acid Treated Gelatin	Base Treated Gelatin	Gelatin
Nitrogen	MRC	4.30	12.7	9.1	-
	TAPPI Monograph 23	5.0	-	-	-
	DuPont	4.75	-	-	-
Air	MRC	3.9	14	-	-
	Lord Rayleigh	-	-	-	761
Helium	MRC	712	115	316	-
	Lord Rayleigh	-	-	-	140,000
	DuPont	766	-	-	-

Swift and Co. has also reported that preliminary data shows diffusion rates are not increased when the films are plasticized, which is an extremely interesting property.

Comparison with Oriented Polymers: The studies of gelatin as an engineering material are admittedly incomplete at this time. Review of data obtained on the films indicate gelatin, an unoriented polymer, compares quite closely with several high strength oriented polymers.

The Table 11 below is presented for comparison purposes.

Table 11

	T°C	Gelatin	Mylar*	H Film*
Tensile Strength (psi)	25	21,000	23,000	20,000
	-	12,000 @ 150°C	5,000 @ 200°C	12,000 @ 200°C
Modulus of Elasticity (psi)	25	1,000,000	500,000	400,000
	200	130,000	50,000	250,000
Break Elongation, %	25	7	100	70
	200	7	Large	70
Moisture Uptake	25	13% @ 40% RH	0.2	1.3 @ 50% RH
Resistance to Organic Solvents		Good	Fair	No known solvent

*E. I. duPont de Nemours & Co. thermosetting development film.

Glass Cloth Reinforced Laminates: Some investigation was accomplished in producing glass-gelatin laminates.

This work involved an attempt to determine the factors important in producing the laminates, such as time-temperature-pressure cycles, cross-linking agents, and surfactants on mechanical properties and water sensitivity. Surprisingly good flexional strengths were obtained in the few laminates made, considering the very limited amount of work done. Optimum press cycle, cross-linkers, and pre-impregnation techniques for the cloth were not really ascertained. Flexural strengths as high as 77,900 psi were obtained with laminates which is in the range of high grade epoxy systems. Future effort should yield even better results. Tables 12 and 13 contain properties of the gelatin films with chemical modifiers and the effects of press cycle variations on the laminates.

It should be noted that the work on laminates was not connected with efforts to produce expandable structures, as they were cured and rigidized during the press cycle. The reported data is included merely as a possible point of interest in discussion of gelatin.

Table 12

Mechanical Properties of Gelatin Containing Chemical Modif

		Tensile Data							
	Yield	Failure	Elongation				Modulus of El		
Description	psi	psi	Yield %	Fail %	25	50	75	100	
G8 Gelatin	18,000	18,300	6	9	>1000	220	120	100	
G8 Gelatin, Vacuum Dried		20,900		4					
G8 Gelatin +									
2.5% Dialdehyde Starch	16,700	17,000	6	9	>1000	450	170	120	
5.0% " "	16,800	18,100	5	11	>1000	160	94	64	
10.0% " "		14,800		6	>1000	680	220	200	
10.0% " "		14,700		4	>1000	400	130	66	
30.0% " "		12,400		3	>1000	110	56	30	
30.0% " "		11,100		2	>1000	600	84	38	
G8 Gelatin +									
2.5% Dialdehyde Starch, Vacuum Dried		18,300							
G8 Gelatin +									
10% Dialdehyde Starch		15,100		4					
10% " "		15,300		3					
30% " "		11,400		2					
(above 3 dried at 100°C)									
G8 Gelatin +									
10% Dialdehyde Starch		16,800		5	>1000	420	260	135	
30% " "		5,200		2					
(Heated 3 hrs 125°C)									
G8 Gelatin soaked CH_2O vapor		17,500		5	>1000	600	160	80	
G8 Gelatin +									
2.5% Dialdehyde Starch		16,000		4	>1000	330	120	78	
5.0% " "		15,600		4	>1000	230	90	47	
(Soaked in formaldehyde vapor)									
G8 Gelatin + 5% Dialdehyde Starch - soaked in NH_4OH vapor		10,500		4	>1000	440	180	100	
G8 Gelatin + aqueous formaldehyde		9,100		3	>1000	88	19	9.2	
G8 Gelatin + aqueous formaldehyde dried at 100°C		12,400		3					
G8 Gelatin + aqueous formaldehyde dried at 125°C		15,600		4	Sample very brittle, broke				

Table 12

anical Properties of Gelatin Containing Chemical Modifiers, Glycerol, and Inert Fillers

Tensile Data			Modulus of Elasticity Data (psi x 10^{-3} T°C)												Flexural Data	
Failure	Elongation														Failure	Inch
psi	Yield %	Fail %	25	50	75	100	110	120	130	150	175	200	225	psi	Deflection
18,300	6	9	>1000	220	120	100	74	42	30	29	56	24	2	17,000	0.09
20,900		4												8,500	0.04
17,000	6	9	>1000	450	170	120	96	66	32	27	42	20	1.2		
18,100	5	11	>1000	160	94	64	45	28	18	20	25	12	2.6		
14,800		6	>1000	680	220	200	160	110	90	120	210	23	<2.0	13,000	0.08
14,700		4	>1000	400	130	66	40	22	12	2.0 @ 135°C					
12,400		3	>1000	110	56	30	20	12	16	7	4.6	3.5	1.6	9,600	0.07
11,100		2	>1000	600	84	38	34	32	26	5	3.2	3.1	1.6		
18,300															
15,100		1													
15,300		3													
11,400		2													
16,800		5	>1000	420	260	135	125	105	84	56	43	25	2.5		
5,200		2													
17,500		5	>1000	600	160	80	58	64	60	31	9	5		[illegible]	0.09
16,000		4	>1000	330	120	78	70	56	47	32	21	13			
15,600		4	>1000	230	90	47	32	15	7	13	19	6			
10,500		4	>1000	140	180	100	56	11	10	10	10	-	2.0		
9,100		3	>1000	88	19	9.2	7.4	5.0	5.0	3.7	6.4	1.1			
2,400		3													
5,600		4	Sample very brittle, broke at start of test.												

Table 12, Continued

Description	Tensile Data: Yield psi	Failure psi	Elongation: Yield %	Fail %	Modulus of Elast: 25	50	75	100	11
G8 Gelatin + aqueous formaldehyde - soaked in CH_2O vapor		8,100		3					
G8 Gelatin + aqueous formaldehyde - soaked in NH_4OH vapor		12,600		4	>1000	80	20	5.4	
G8 Gelatin + aqueous formaldehyde - soaked in NH_4OH vapor, dried at 100°C		9,000		2					
G8 Gelatin + hexamethylenetetramine	12,300	11,700	4	13	1000	320	51	14	1
G8 Gelatin + hexamethylenetetramine - soaked in CH_2O vapor		14,800		5					

12, Continued

Table 12, Continued

Tensile Data			Modulus of Elasticity Data (psi x 10^-3 T°C)													Flexural Data	
Failure	Elongation															Failure	Inch
psi	Yield %	Fail %	25	50	75	100	110	120	130	150	175	200	225	psi	Deflection
8,100		3													
12,600		4	>1000	80	20	5.4	3.8	3.2	2.8						
9,000		2													
11,700	4	13	1000	320	51	14	11	11	10	6.4	4.5	4.5	-		
14,800		5													

Table 13

Effect of Press Cycle on Laminates

Sample	Press Cycle Time (min.)	Temperature (°C)	Pressure (psi)	Flexural Strength (psi)
PC1	20 30 90	102 105 115	0 50 100	55.400
PC2	5 25	135 135	0 100	53.700
PC3	20 30 60	102 105 135	0 50 100	67,200
PC4	5 55	135 135	0 100	57.400
PC5	20 30 60	102 105 150	0 50 100	48.700
PC6	20 30 60	102 105 135	0 50 150	57.300
PC7	20 30 60	102 105 135	0 50 100	69.600
PC8	20 30 60	102 105 135	0 50 1000	77.900
PC9	30 60	105 135	0 1000	64.400

III. GELATIN RIGIDIZED STRUCTURES RESEARCH

Concept: In November of 1962, Mr. Forbes, co-author, initiated research directed toward determining the feasibility of utilizing gelatin materials as a rigidizing medium for expandable support type structures. Structures were fabricated from gelatin plasticized films or gelatin-fiberglass composites, the plasticizer, a glycol, permitted the folding or packaging of the structure. These structures, after deployment in the vacuum of a space chamber, rapidly rigidized by the migration or evaporation of the plasticizer. In reality the gelatin is dehydrated during rigidization.

Preparation of Gelatin Film Structures: Gelatin cast film structures and sprayed film gelatin structures were prepared by dissolving about 20% to 30% by weight of gelatin dry crystals into 70% to 80% by weight water at a temperature between 120°F. to 140°F. This solution is then either cast into films or sprayed into sheets. The sprayed or cast films were air dried for about one to two days or until the films had dried to a non-tacky rubbery state. These films were then sealed into a plastic air tight bag and stored at 33°F. When these films were removed from containers and exposed to a vacuum the remaining water, about 20% to 40% by weight, rapidly migrated and caused the film to rigidize. Plasticizers, such as ethylene glycol, sorbitol, glycerine, and polyethylene glycol can be used in place of water. Utilizing these plasticizers will permit lower plasticizer concentrations.

Gelatin fiberglass composites are prepared by making a gelatin-water or gelatin-water-plasticizer solution and pouring this solution over the fiberglass cloth or dipping the glass cloth into the solution. Once the cloth laminate is thoroughly saturated, 25% gelatin by weight minimum, the laminate is placed in a temperature controlled hydraulic press with a platen pressure of 25 psi to 100 psi and a platen temperature of 140°F. to 200°F., and the laminate is cured for about 20 to 30 minutes. Then the press is allowed to cool to 60°F. for about 20 to 30 minutes. The resulting laminate panel is placed in an air tight container until usage. It has been found that a better bond between fiberglass and gelatin can be achieved by adding 1% by gelatin weight of a liquid detergent to improve the wetting characteristic of the gelatin solution. Addition of 1% by gelatin weight of a cross-linking agent previously discussed will produce a water proof laminate; however, it is recommended that the laminates be cross-linked by exposing the laminate to a vaporized cross-linking agent. This recommendation has been made because the addition of a cross-linking agent tends to prevent adequate bonding between gelatin and fiberglass which in turn causes delamination failures. Two individual sheets of fiberglass gelatin may be joined together by a simple lap joint technique or gelatin pressure bonded tape. The addition of a cross-linking agent also will interfere with ability to achieve a satisfactory structural joint.

Fabrication and Testing of Expandable Gelatin Structures: The first attempt at fabricating a gelatin structure was made by brushing the gelatin solution over an inflatable form with fiberglass cloth reinforcing laid over

.he form. This inflatable form was a 24 inch diameter dome with a six inch cylinderical skirt. After fabrication this structure was folded into a compact package and stored for ten days after which it was placed in a vacuum chamber. The chamber was evacuated to 10^{-1} Hg., for four hours. Once, the chamber had been evacuated, the structure was pressurized to an internal pressure of 200 mm Hg. Radiant heat lamps installed in the test chamber simulated solar radiation which provided a structure surface temperature of about 100°F. The resulting structure was quite rigid for its weight, but somewhat wrinkled in appearance. The rigidized structure had a wall thickness of about thirty mils. Figure 3, shows photographs of the expanded rigidized structure.

The next expandable structure was fabricated by spraying the gelatin solution over a rigid form. Once the completed structure had air dried for one day, it was removed from the mold. The sprayed-on gelatin material was quite rubbery and was folded up and stored for a week, after which the structure was inflated to 240 mm Hg., pressure which was quite remarkable since, this structure had no internal air tight membrane, as in the previous experiment. This structure was left in a chamber at 10^{-1} mm Hg., for two hours and thirty minutes, resulting in a rigidized dome of three-eighths inch wall thickness. Figure 4, shows photographs of this dome unexpanded and in the rigidized state. The next major change in fabrication procedure was the incorporation of an airless spray gun for spraying the gelatin. The purpose of the airless spray gun was to eliminate the entrapment of air caused by conventional spray guns. The airless spray gun reduced the amount of entrapped air by 50%. Similar structures, as previously described, were fabricated and rigidized at vacuums of 10^{-1} mm Hg., or less.

After successfully rigidizing a number of small structures the next goal of this research program was the rigidization of a 12.6 foot diameter sphere. A rubberized nylon inflatable sphere was spray coated with a gelatin-1% formaldehyde-water solution, as shown in Figure 5. This sphere was placed in a vacuum chamber which was evacuated to 10^{-1} mm Hg. The sphere was internally pressurized to about .5 psi; however, shortly after pressurizing the sphere shrinkage cracks occurred. Gelatin coatings, as previously discussed, have a tendency to shrink during migration of the plasticizer while the internally pressurized bladder does not, thus the gelatin rigidized coating cracked into approximately three feet by three feet sections.

A re-evaluation was made of this program which resulted in a re-emphasis of gelatin reinforced fiberglass structures. A series of small 12 inch diameter domed test panels were fabricated and rigidized under 10^{-1} mm Hg. The utilization of the fiberglass reinforcing has eliminated the problems of shrinkage cracks and wrinkles in the structural membrane. Several test structures have been fabricated with a brown Kraft paper inner liner integrally bonded to the gelatin-fiberglass composite for the purpose of reducing porosity or pin hole size leaks. A pillow type gelatin fiberglass structure was fabricated from two panels two feet by three feet with Kraft paper inner liner.

Figure 3, Expanded Rigidized Gelatin Fiberglass Structure

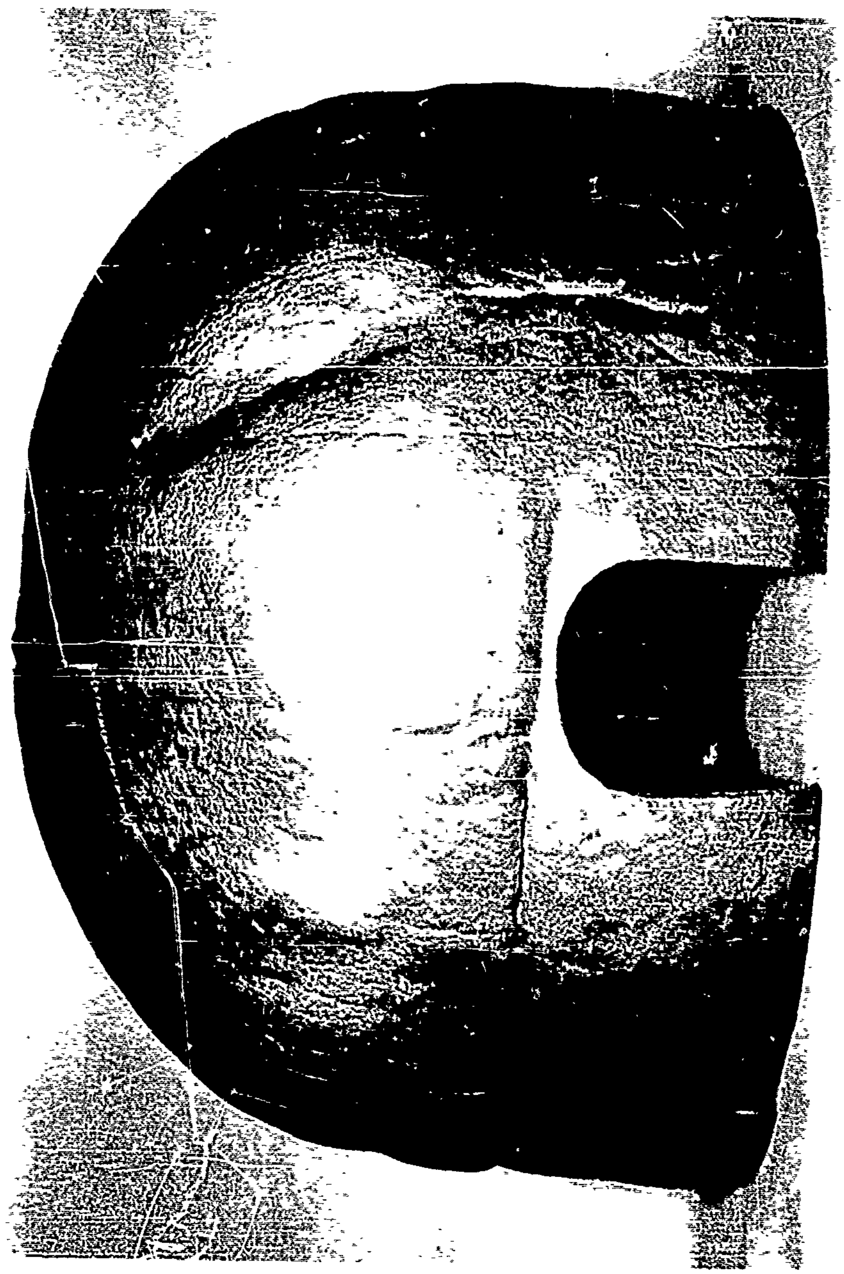


Figure 4, Gelatin Unreinforced Unexpanded and Expanded - Rigidized Dome

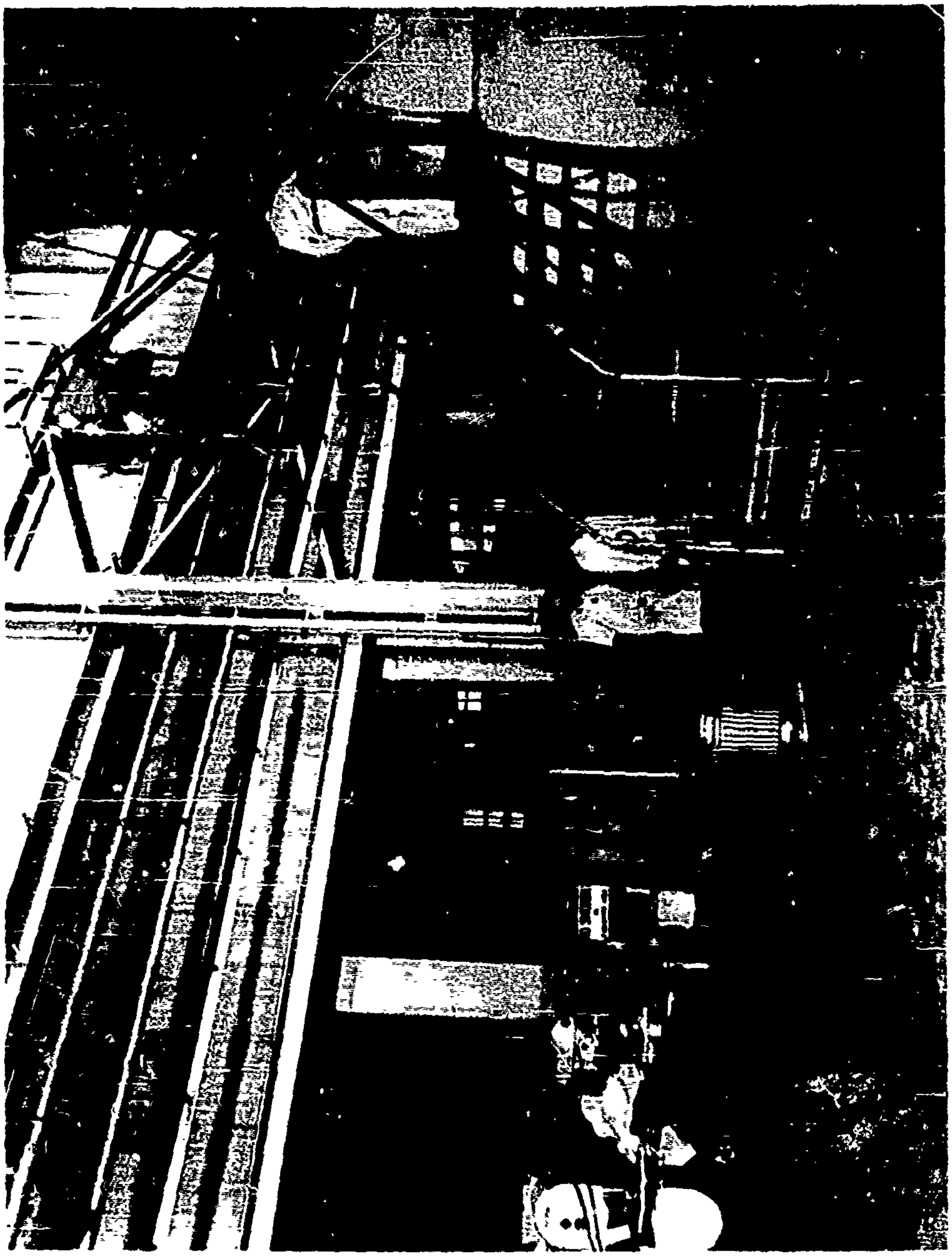


Figure 5, Fabrication of 12.5 ft. diameter Gelatin Sphere

These two panels were seam welded around the perimeter. This structure was plasticized with 25% ethylene glycol. This structure was rolled up and stored in a 3-1/2 inch x 20 inch long cylinder, later it was placed in a vacuum chamber and was rigidized within three hours at a vacuum of 10^{-1} mm Hg. This structure was very smooth and no shrinkage cracks were evident.

An expandable pyramid, three feet on a side was made of gelatin fiberglass. This pyramid was rigidized under a 10^{-1} mm Hg., vacuum condition in two hours. Figure 6 shows photographs of this pyramid unexpanded and expanded.

After the latter experiment, it became apparent that the impregnating process utilized for applying the gelatin to the fiberglass cloth was ineffective. Therefore, the Monsanto Research Corporation was given a contract by the Air Force Aero Propulsion Laboratory to preimpregnate continuous rolls of 18 inch wide fiberglass cloth. The following is a copy of the impregnation procedure used by Monsanto:

PROCEDURE FOR PREPARING PLASTICIZED GELATIN PREPREG

The addition of plasticizing agents such as the polyhydric alcohols (glycerine, ethylene glycol, propylene glycol, etc.) to the gelatin-glass cloth prepreg will produce flexible laminates. The flexibility of these laminates is due to the plasticization of the gelatin by the plasticizer only in part. The water present in the laminate appears to be the main plasticizing agent. The polyhydric alcohols, besides serving as plasticizers, also act as hygroscopics to hold the water in the gelatin. Consequently, the moisture content is an important factor in preparing a flexible laminate. Tests to date show that glycerine is the best plasticizing agent.

Several advantages are gained when a plasticizer is used. The gelatin solution is easier to prepare. The plasticizer aids in dissolving of the gelatin. Also the prepreg is flexible and easy to handle.

The procedure for treating and pressing the plasticized gelatin laminates is as follows:

I. MIXING

A. To 75 parts of warm water (60°C.) add .65 parts of preservative (67.5 parts ethyl phydroxybenzoate in 200 parts acetone). Mix thoroughly while adding.

B. Mix in 10 parts of plasticizer (40% glycerine based on gelatin).

C. Add 25 parts gelatin. Mix thoroughly, breaking up any lumps.

D. Store several hours at 60°C. to deaerate.

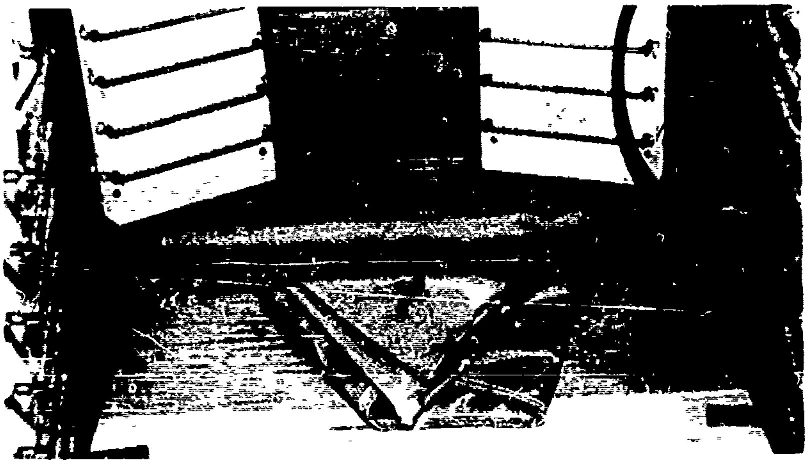


Figure 60, Unexpanded Gelatin Fiberglass Pyramid

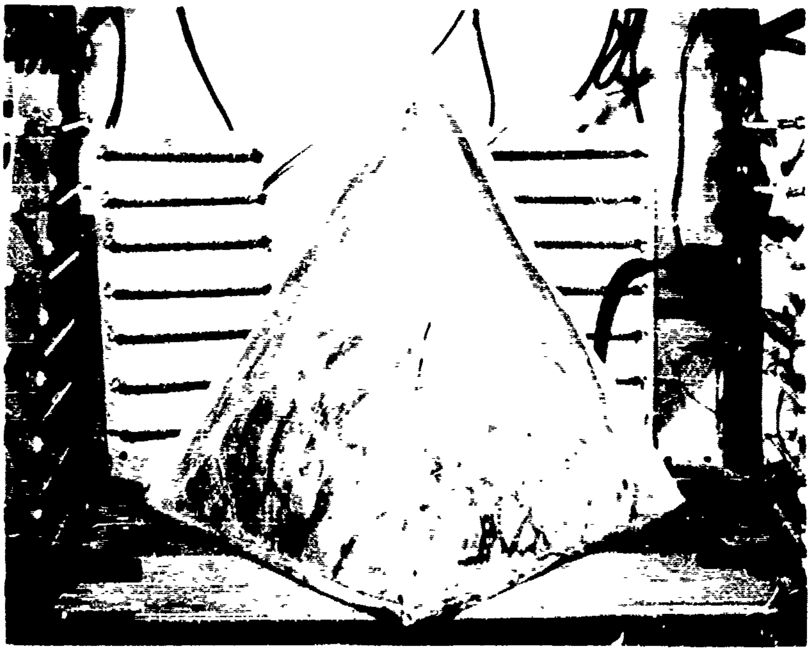


Figure 61, Expanded Rigidized Pyramid

II. PREIMPREGNATION

A. Glass Fabric E-181, A-1100 size is used.

B. Three zone oven and dip pan of the fabric treater are heated to proper temperature.

(1)	First zone	-	90°C.	(Zone length is 3 ft.)
(2)	Second zone	-	110°C.	(Zone length is 3 ft.)
(3)	Third zone	-	200°C.	(Zone length is 3 ft.)
(4)	Dip pan	-	55°C.	(Fabric immersed at least 6 in.)

C. Variable speed drive is adjusted to give 20 inches/minute.

D. Dip solution is added to dip pan and treater is started.

E. Continue impregnation and drying until desired amount is obtained. Test material occasionally.

III. TESTING

A. % Resin

(1) Periodically cut a swatch of cloth 16" x 12".
(2) Weigh to nearest tenth of a gram.
(3) Compute resin pickup (weight of untreated cloth this size is 35.5 grams).
(4) Raise or lower temperature of dip pan to control resin pickup. Maintain 40% resin.

B. Volatile Content

(1) Use the piece of material cut for the % Resin Test.
(2) Weigh and dry in oven 15 minutes at 125°C.
(3) Weigh and compute water loss.
(4) Maintain volatile loss at 2% by adjusting speed of cloth through oven. Prepreg must not become too sticky (determined by touch) or it will stick to itself on the roll.

IV. PRESSING

A. Cut prepreg to desired size

B. Build up press pack as follows:

(1) Plate
(2) Eight (8) laminations of Kraft paper
(3) Polished plate
(4) Mylar sheet
(5) Prepreg laminations
(6) Mylar
(7) Polished plate
(8) Insert thermocouple in 2 inches from edge
(9) Eight (8) laminations of Kraft paper
(10) Plate

C. Load in press and close to platen pressure

D. Turn on steam and adjust to obtain and maintain 135°C.

E. When temperature reaches 135°C., apply 200 psi on laminate.

F. Press at 135°C., for 20 minutes.

G. Cool to 25°C., relieve pressure, and remove laminate from press.

V. CONDITIONING

A. Condition laminates at 100% R.H. for maximum flexibility.

Recommendations: The authors make the following recommendations:

1. Investigate the feasibility of utilizing gelatin to rigidize expandable core structures.

2. Continue research to characterize the properties of gelatin.

3. Investigate the combining of gelatin with other proteins, such as soy protein.

4. Develop a reactive plasticizer system for gelatin which on exposure to a vaporized catalyst will rigidize it. This development would provide a gelatin rigidization system which would be useful for both space and ground applications.

"MICRO ENCAPSULATED POLYURETHANE FOAM REACTANTS FOR AEROSPACE STRUCTURES"

By Robert E. Kass and Lawrence R. Lankston

The National Cash Register Company, Dayton, Ohio

Prepared for the
"Conference on Aerospace Expandable Structures"
22, 23, 24 October 1963
Sugar Camp, Dayton, Ohio

This paper consists of two parts. The first part is concerned with the chemistry of capsular polyurethane foam systems and the second part with the application of these systems to the engineering and development of rigidizable shelters and solar collectors.

The Chemistry of Capsular Polyurethane Foaming Systems

(1) The National Cash Register Company became interested in the rigidization of aerospace structures through the development of techniques for applying micro-encapsulation technology to polyurethane reactants. This technology provides a method for separating reactive components, such as those used in the formation of polyurethane foams. Both liquids and solids can be encapsulated. Capsular liquids exhibit the behavior of a pseudo solid. Figure 1 shows two examples of encapsulated polyurethane reactants. The material on the right is an encapsulated polyol. The other material is an encapsulated solid polyisocyanate. Capsule sizes can vary from a few microns to as large as a quarter of an inch. A major advantage of encapsulated foam reactants is their ability to be premixed, without reaction, for prolonged periods.

This paper will not discuss the NCR micro-encapsulation technology or the various coacervation processes. Rather, the paper is confined to a discussion of the development of capsular urethane foaming systems for use in an aerospace environment.

The general reaction involved in the preparation of polyurethane foams is the reaction of a polyisocyanate with a polyol to form a urethane linkage. Certain co-reactants when used with the proper amount of catalyst, surfactant and blowing agent, will produce a rigid foam. Standard commercial foams usually involves the use of carbon dioxide or freon

blowing, however, in high vacuum environments these blowing mechanisms are not operable and new ones must be devised.

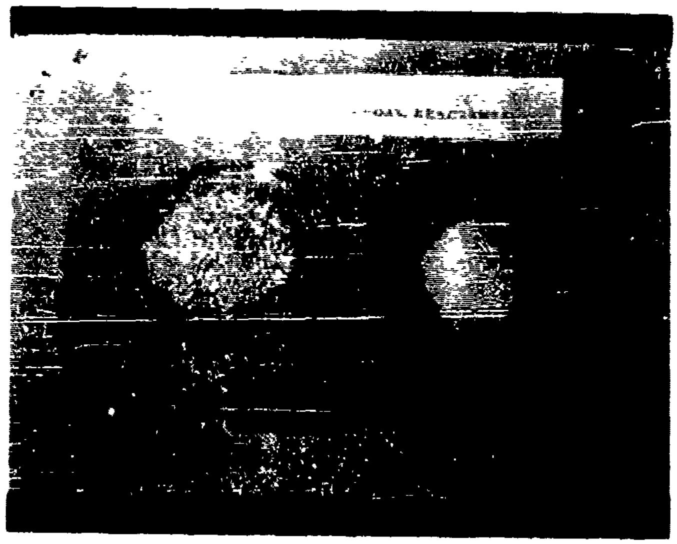


Figure 1. Encapsulated Foam Reactants

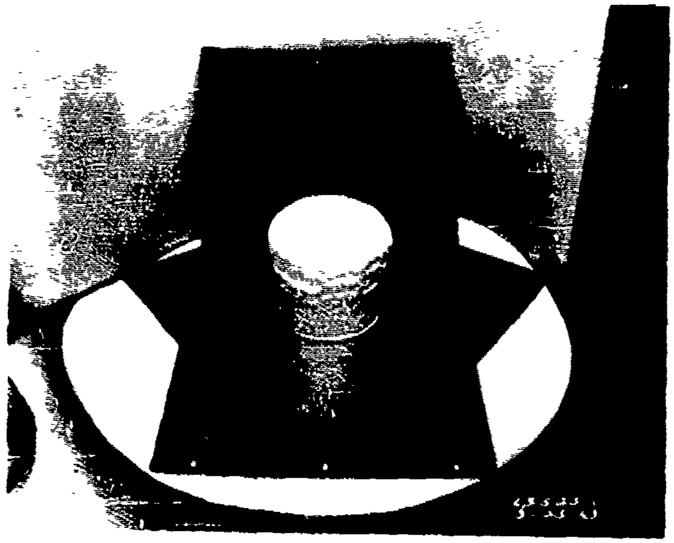


Figure 2. Commercial Urethane Foam-
Prepared at Atmospheric Pressure

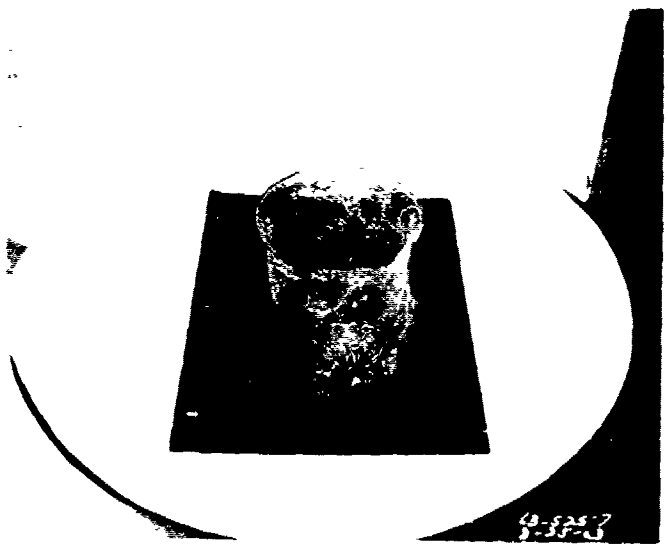


Figure 3. Commercial Urethane Foam Materials
Reacted at One Millimeter Pressure

Figure 4. Sample of Foam From Capsular Atmospheric Formulation

The type of foam produced at atmospheric pressure, using a standard commercially available, rigid foam formulation (freon blown) is shown in Figure 2. This foam has a density of approximately two pounds per cubic foot. When this formulation is reacted in a vacuum environment, at about one millimeter pressure, a product as shown in Figure 3 results. The difference in properties of Figure 2 and Figure 3 shows that the lack of pressure increases the volume occupied by the blowing gas, thereby causing too rapid a blowing rate and foam collapse. This dramatizes the problem and points out that basic formulation studies involving polyurethane reactants were required before autogenous urethane systems possessing the ability to be used in high vacuum environments could be developed.

To test the concept of urethane foams, using encapsulated ingredients, the initial experiments involved an atmospheric foam formulation. The formulation studied consisted of a prepolymer isocyanate resin (Urefoam R) which could be foamed and cured with water. Dabco (amine) and T-9 (tin compound) were used as catalysts. The water source was encapsulated, hydrated sodium borate, which dehydrates upon gentle heating. It should be noted that in all cases during these investigations thermal energy was utilized to activate the capsular foam formulations. Upon heating, the material inside the capsule was released and allowed to react with external or other capsular materials. A cross section of foam sample, made using the capsular Urefoam - R system, is shown in Figure 4. The cell structure is similar to a typical atmospheric formed polyurethane foam.

After establishing the feasibility of the capsular concept, vacuum foaming systems and the capsular means to translate these systems to stable, one package units, were developed. As part of this overall work the vacuum volatility of a number of polyurethane ingredients was examined. Since the efforts were directed toward systems which could be stored at low pressures until heat activated, materials which would not be too volatile during prolonged storage, were required.

Some of the materials that were tested for volatility at reduced pressure are shown in Table 1. The data was obtained on the un-encapsulated reactants. The weight loss in the 625 material was thought to be free TDI which is commonly found in the quasi-prepolymer resins. The higher molecular weight and more completely prereacted Urefoam R did not show continued loss after the initial volatiles were removed. Materials that could be stabilized to a constant weight with small weight

loss were considered useful in the program. The PAPI (polymethylene polyphenylisocyanate) isocyanate was the least volatile of the liquid polyisocyanates. Most of the polyols examined showed low weight loss at room temperature. Materials such as glycerine, which showed relatively high volatility at room temperature, were excluded from the acceptable materials list.

The first formulation that was developed for low pressure use involved the liquid polyisocyanate PAPI and an encapsulated polyol, trimethylol propane. The blowing mechanism used for this system was based on thermally releasing the trimethylol propane, which in turn reacted with the Hydripill to produce hydrogen. Hydripill is a trade name for a mixture of sodium borohydride and cobalt chloride. Catalysts and surfactants were added as required.

Figure 5 is an example of the product obtained from the PAPI system. This foam was produced at 10^{-3} Torr and possesses a density of 8 pounds per cubic foot, plus a tensile strength of 100 psi. It was found, however, that in going to lower density foams in this system, the product became somewhat friable. For this reason it was necessary to develop additional foam formulations which possessed better physical properties.

Development of a foam formulation which possesses the required physical and structural properties was based on the use of capsular methylene diphenyl diisocyanate, polyol, surfactant and aluminum pigment. This particular foam formulation has the appearance of a viscous slurry or paste. Figure 6 represents the type of foam obtained from this formulation. The foam on the right was produced at 10^{-6} Torr without pigment. The one on the left is the same formulation, but with aluminum pigment added. This pigment was found to contribute to the

nucleation requirements of the foaming reaction, plus increasing the transfer of the formulation leading to a more uniform cure and a stronger foam.

This system has proven operable from 1 Torr to vacuums higher than 10^{-6} Torr with the same degree of success.

Vacuum Stability of Urethane Foam Components at 25 Deg. C.

Material	Pressure	Time in Hrs.	% Weight Loss
Polylite 625	10^{-5}	2	2.8
"	"	7	6.2
Polylite 505	"	2	0.3
"	"	7	1.1
Urefoam R	"	2	2.0
"	"	4	2.0
Niax D22	"	2	0
T-9	"	2	0
PAPI	"	2	0
TMP	"	2	0.3
MDI	"	2	0
DC0113	"	2	1.0
TDI	10^{-2}	2	3.8
Glycerine	10	1	19.0

Table 1. Vacuum Stability of Urethane Foam Components

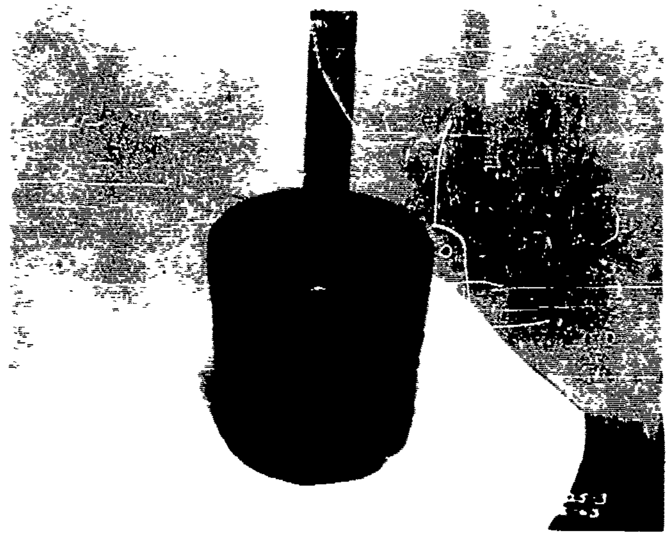


Figure 5. Piece of Foam Rigidized Cryogenic Pipe

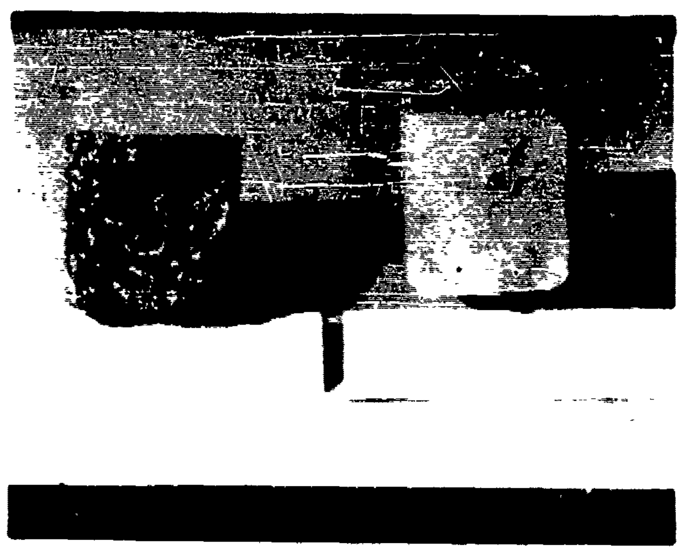


Figure 6. Examples of Foam Prepared From Capsular Formulation at 10^{-6} Torr

The Application of Capsular Foams

As soon as consistent success was obtained with the slurry type foam formulations, engineering work involving the problem of utilizing the foam for the fabrication of a 7 foot diameter by 7 foot high space shelter was initiated. Before work on the structure could be performed, the tensile strength of the foam had to be determined. From literature and data that had been obtained during the research phase of the contract, it was known that the density of the foam affected its ultimate tensile strength. In addition, the capsular foam cell sizes were large compared to atmospheric polyurethane foams. This large cell size dictated the use of specimens having large cross sectional areas for tensile testing. It was determined that the tensile specimens must have a minimum cross section of four square inches before consistent test results could be obtained from the same bulk sample of foam. This indicated that tests run on smaller sections test only the strength of the cells and voids instead of the foam strength. By these tests it was determined that the tensile strength of the foam was not significantly influenced by the vacuum range (1 Torr or at 10^{-6} Torr) in which the foam was prepared. All property testing to date has been performed under average atmospheric laboratory conditions and no attempt has been made to test the foam in a vacuum or at other than ambient temperatures.

Figure 7 indicates that a foam having a density of 2 pounds per cubic foot will have an ultimate tensile strength of less than 10 pounds per square inch, while a foam having a density of 5 pounds per cubic foot will have an ultimate tensile strength of 40 pounds per square inch. By contractual requirements, NCR has worked only in the 2 to 5 pound per cubic foot range. All of the data presented in this paper represents the strength of the NCR slurry type vacuum foam without the addition

of strengthening agents.

Figure 8 shows the maximum wall stress developed in a 7 foot diameter, thick walled cylinder under 15 pounds internal pressure versus wall thickness. The curves are asymptotic at about 15 pounds per square inch as expected. If foam is utilized as the sole structural element of the cylinder, a weak foam will require a very thick wall. In the densities we have been restricted to, a 5 pound per cubic foot foam developing 40 pounds per square inch will require a wall thickness of 12 inches or more. This is rather thick and out of the question for a practical shelter. However, such a shelter will show the feasibility of fabricating foam into standard structural shapes. A 15 pound per square inch pressure is rather high as far as structural loads are concerned, (a typical office floor load is seldom as high as 1 psi), an air-inflated, rigidized cylinder can be achieved by using a foam with a higher working stress or by building a composite wall consisting of foam and a stout cloth-like material. The composite approach does not use the foam to obtain structural strength. The foam is used to rigidize the cloth bag in case of loss of internal pressure.

Figure 9 shows a cross section of the shelter wall. Vertical and horizontal cords are noted in the figure. These cords are used to develop the stress required to maintain a flat floor. A flat floor is one of the problems which developed while trying to make a cylindrical space shelter. A cylinder with a flat floor is very difficult to make in one operation using air inflated structures. For this reason, the approach has been a two step operation; the floor is first foamed and cured, followed by the foaming and curing of the sidewalls. The horizontal cords are used during the foaming of the floor and extend through the exterior bag to a toroid which surrounds the base of the shelter. The toroid is inflated

and the horizontal cords pulled taut, holding the interior floor flat while the foam is cured. After the foam floor is cured the toroid is no longer used, but the vertical cords have been attached beneath the foam floor and when internal pressure is applied to the shelter, the vertical cords transmit the load into the floor preventing the bottom of the interior bag from becoming concave. All work to date has been carried out on quarter scale engineering prototypes. It was decided that all information of flow, placement of foam, use of horizontal and vertical cords and use of a toroid could be determined on quarter scale models, thereby eliminating the expenditure of several full scale models during development work.

Figure 10 shows an external view of a quarter scale engineering prototype which was foamed at 1 Torr. The penetration was placed in the model after the foam was rigidized. Figure 11 shows a cross section of this quarter scale model and shows the smoothness of the interior surface. The sidewalls were of uniform thickness and were perpendicular to the floor. The floor has a slightly convex contour, but this is in the direction of a more rigid floor and was considered acceptable. Cords are shown coming from the floor upward onto the wall. The cords are attached to the wall with a heat sensitive tape and are the vertical cords referred to in Figure 9.

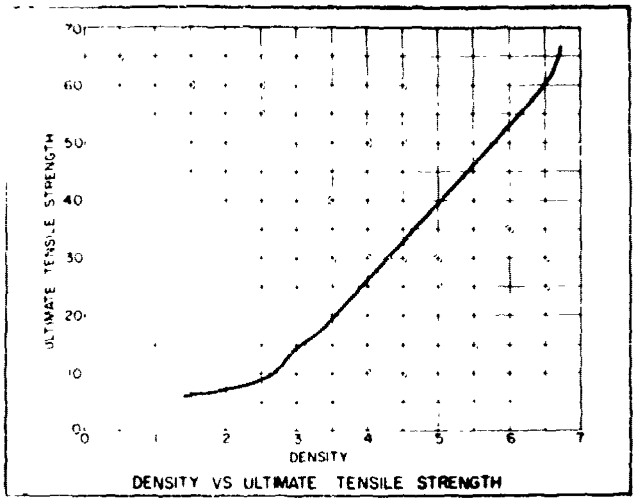


Figure 7. A Graph of Density Versus Ultimate Tensile Strength of NCR Vacuum Foam

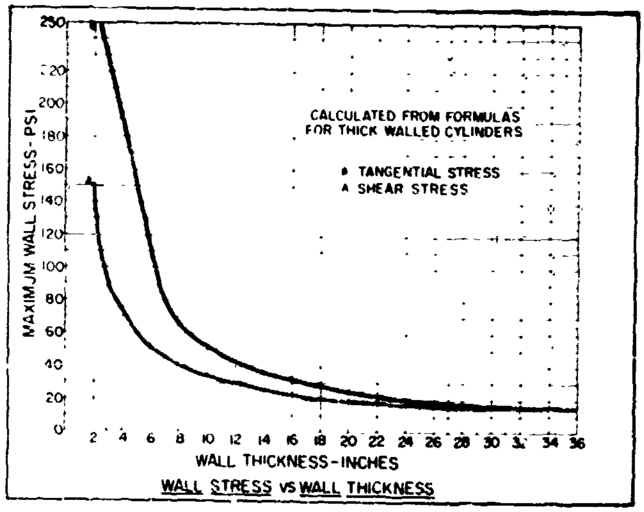


Figure 8. A Graph of Wall Stress vs Wall Thickness for 7' Diameter Thick Walled Cylinder

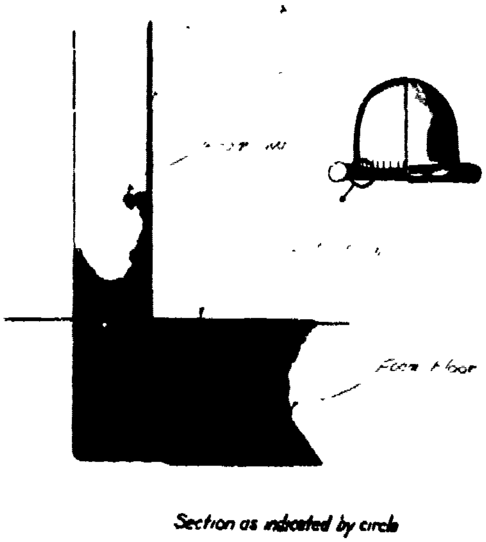


Figure 9. **Proposed Cross Section of a Space Shelter**

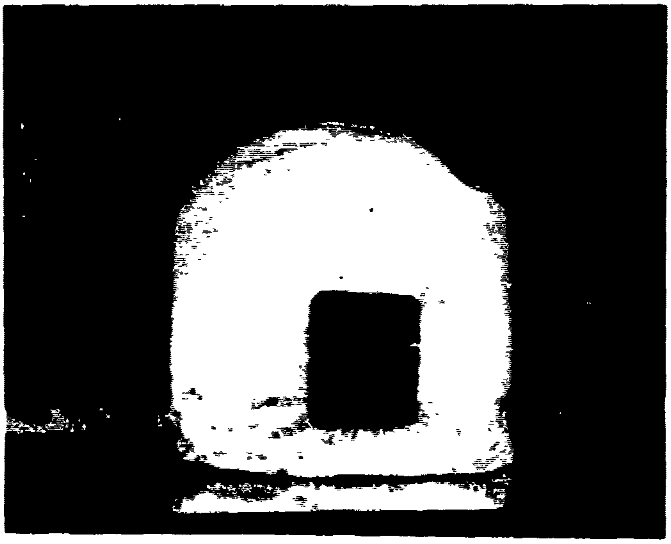


Figure 10. **One-Quarter Scale Model of a 7' Space Shelter Foamed at 1 Torr**

Figure 11. Cross-Sectional View of a One-Quarter Scale Shelter Model Foamed at 1 Torr

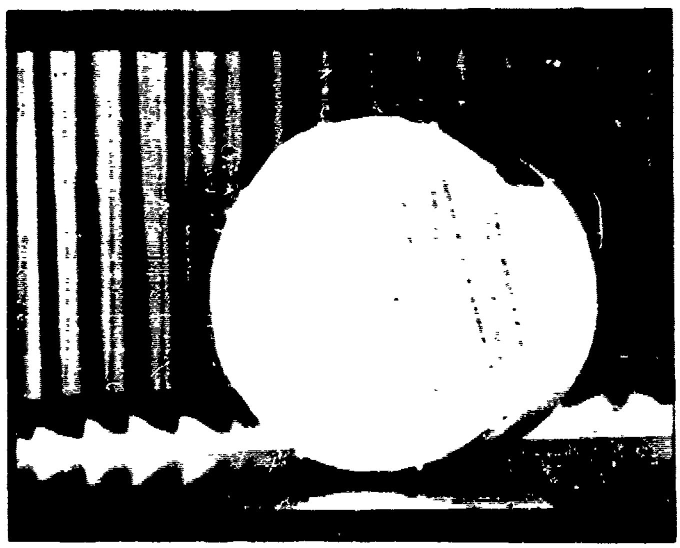


Figure 12. Location of Heating Element on Floor of Quarter Scale Shelter Model

Figure 12 shows the placement of the heaters on the bottom of the quarter scale shelter. The heaters are 0.001 inches thick flexible nichrome tape by one-quarter inch wide and occupy 10% of the floor area. The energy required to activate the foam in the floor was 0.4 Btu per pound per $^{\circ}$F. These heaters are placed directly on the Mylar and carry a current of 3 amperes during foaming. There has been no problem with the heaters burning the Mylar during the foaming and curing operation. The heaters are flexible and can be folded and unfolded several times without damage.

Figure 13 is a cross section of one-quarter of a slab of foam containing aluminum powder and was prepared at 1 Torr. It was found that aluminum powder increased the strength of the vacuum foam by a factor of 2, bringing the strength of a 3 1/2 pound per cubic foot foam up to 45 pounds per square inch. While 45 pounds per square inch still is not adequate to reduce the wall thickness of the shelter to 4 inches, it points up the fact that perhaps a more dense foam, in the range of 8 to 10 pounds per cubic foot, with aluminum powder, may have sufficient strength to reduce the wall thickness to a reasonable value.

During work with the quarter scale models and foam samples for tensile testing, further information was gathered on the vacuum foams and also on the behavior of Mylar. Figure 14 shows the temperature history of a foam prepared at 10^{-6} Torr. The heaters were energized at the beginning of the test for 40 minutes. The rise in temperature after the heaters were de-energized is due to the exotherm of the foam and it will be noted that the maximum temperature reached was less than 120°C. The foam was then allowed to cure in the vacuum and cool to a temperature of 50°C before the pressure was increased and the

sample removed for further testing. Recent tests with formulations designed for lower exotherms have lowered this maximum temperature to less than 100°C. The tensile strength of the low exothermic foams is about the same as the higher exothermix formulations. The total expired time, from initiation of the reaction to cure is approximately one hour.

Figure 15 indicates the elongation of Mylar under various loading conditions versus time. The Mylar was cut into one inch wide strips, 100 inches in length and was suspended from the ceiling. To date, the strips have been tested in excess of 1200 hours and the curves are extending as might be extrapolated from the graph, indicating that the Mylar has a continuous elongation under load. NCR is presently evaluating the use of a coating between the Mylar and the predistributed foam which will be used on the solar collectors to reduce this steady growth. It is apparent that with a material such as Mylar, the continuous creep makes it very difficult to produce an accurate contour as the Mylar has a varying creep rate versus temperature and versus load. Using simulated 2 foot diameter solar collectors, which are made by placing flat sheets of Mylar over a fixture and coating these simulated collectors with an intermediate coating, the change in length has been greatly reduced and in some cases completely eliminated. The foam has then been applied and rigidized with practically no elongation. This is quite an improvement over what is achieved with the foam in direct contact with the Mylar. Work is presently being directed towards optimization of the thickness of the backcoating and to the determination of whether or not a flexible coating can be used on the collector. It is felt that the slurry type formulation is an intermediate formulation and that a totally "dry" formulation can be made and predistributed over the surface of the object to be foamed. This formulation will have a fast reaction time and will grow outward from the surface to be foamed, eliminating the problems encountered with a "flowing" foam.

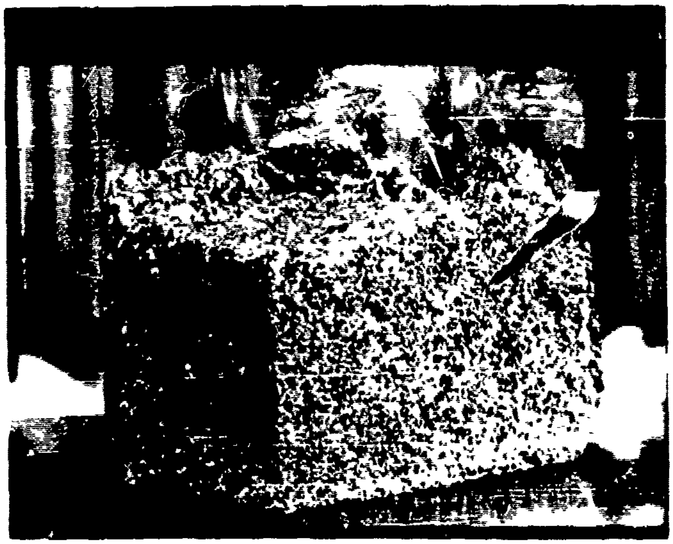


Figure 13. Cross Section of 15 Pound Block of Foam

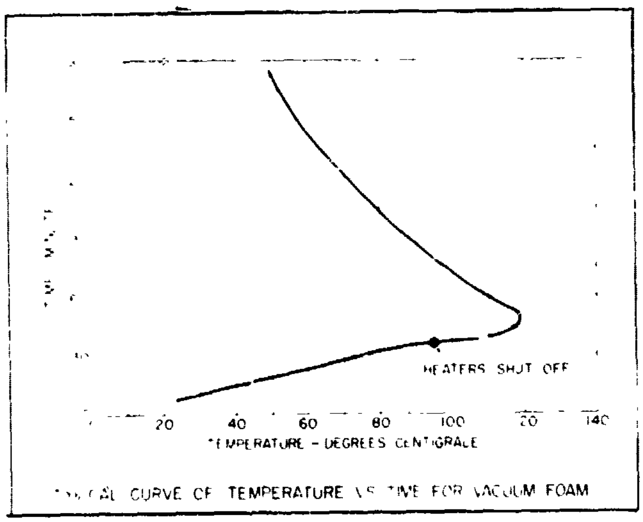


Figure 14. Typical Curve of Temperature vs Time for a Vacuum Foam Reaction

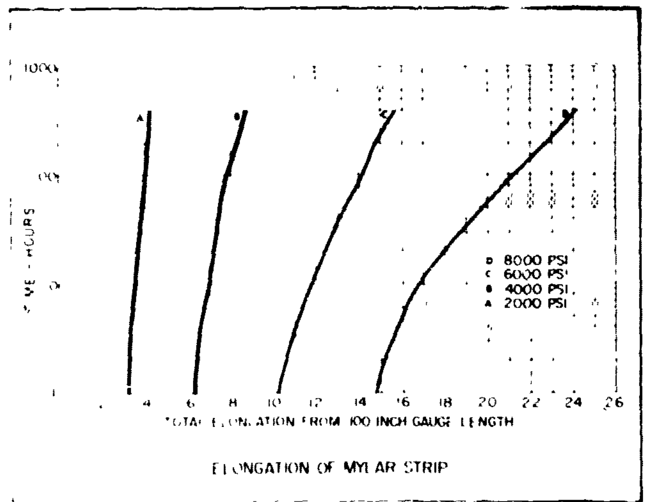


Figure 15. Typical Curves Showing the Creep of Mylar vs Time

Summary

The concept of encapsulating the reactive constituents of a foam formulation has been developed and proven feasible. Rigid polyurethane foams have been produced in a vacuum as high as 10^{-6} Torr; these foams are especially formulated for high vacuum applications. Standard atmospheric formulations will not produce satisfactory foams when reacted in a high vacuum. The encapsulation technique allows premixing and storage of the formulation at room temperatures for long periods of time. Work is presently progressing towards a total capsular foam system in which all of the active ingredients of the formulation will be in a "dry" state suitable for predistribution over the surface of the object to be rigidized. The reaction will be triggered when desired and the fast reacting foam will rise outward from the precoated surface.

In order to reduce the orange peel effect of the foam on Mylar and to reduce the creep of Mylar, a backcoating is being developed. The present coating is rather thick, but it is anticipated that future coatings may be less than 0.005 inches thick.

The work described in this paper was performed under the sponsorship of The United States Air Force Aero Propulsion Laboratory under Contract Number AF 33(657)-8961. The Air Force Project Engineers on this contract are Mr. Fred Forbes and Lt. Ian Thompson.

RIGIDIZATION OF EXPANDABLE STRUCTURES VIA GAS CATALYSIS

T. L. Graham and R. G. Spain

Air Force Materials Laboratory, Research and Technology Division

INTRODUCTION

A variety of schemes have been conceived for the attainment of expandable structures which will automatically rigidize or by other means assume and retain an expanded configuration when deployed in an aerospace environment. Many of the concepts under investigation to attain this common goal involve chemical or physical processes and combinations thereof which in one respect or another are dependent or take advantage of the conditions existing at altitudes high above the earth's surface. Processes being investigated whereby rigidization is physically induced are exemplified by the stressing of thin metallic foils to a curved configuration, the vacuum volatization of plasticizers which are contained in stiff polymeric coatings to render them flexible and the thermal restoration of a material's designed configuration which was induced by crosslinking it in the expanded state and restrained in the packaged or compressed state by refrigeration. Light and heat sensitive polymeric materials which can be converted by radiant energy into solid or rigid cellular products are among the chemical systems being considered. Other than the above, schemes are under investigation which rely entirely upon mechanical principles. Telescoping rigid members and spring loaded rigid frame networks covered with flexible membranes which lock into position when released are approaches which have received considerable attention.

Many uses have been proposed for expandable structures which can be rigidized after being inflated in space. The degree of rigidization required varies with the mission which the structure is intended to fulfill. Structures deployed to collect and store solar energy or to receive and transmit communication waves need only be rigidized to an extent that the combined forces exerted by the radiant energy of the sun and the residual gas present (which do not amount to 1 mm Hg at altitudes of 100 miles) will not collapse them when their inflating gas escapes. Concepts resulting in rigidized structures capable of withstanding large internal pressures could be utilized to establish manned space satellite systems. Structures rigidized to a degree capable of withstanding large external forces would be useful for inducing drag on reentry flights.

More than three years ago a research program was initiated to investigate a chemical concept of rigidization which differs from those previously described in several respects. It is primarily distinguished from other chemical processes by virtue of the fact that the gas employed to inflate the structure also reacts with the polymeric impregnant converting it into a tough rigid crosslinked polymeric product which binds the layers of reinforcing fabric

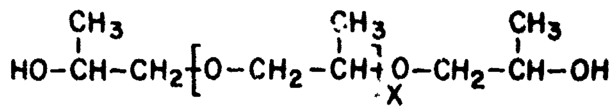
into a solid integral unit. As the reactants are not combined until it is necessary to effect rigidization the problems associated with thermally sensitive chemical mixtures are avoided. No significant weight losses are encountered using this scheme as the components employed are not loaded with volatile materials which are intentionally vented to escape in a vacuum environment. While the various gas catalyzed rigidization systems are dependent upon temperature as to the rate of the rigidization reactions no high threshold temperatures are involved. Structures based on this concept of rigidization are comprised of three components. A high strength fabric coated with an impermeable flexible polymer serves as the cover. It defines the shape of the structure, confines the inflating gas and determines the bursting strength of the unrigidized structure. The ply or plies of reinforcing fabric which are impregnated with the rigidizable resin are intermediate. To prevent the resin impregnated substrate from touching and adhering in the packaged state the structure is lined with a film or coated fabric which is permeable to the gaseous curing agent. This component also forces the structural components together on inflation by transmitting the pressure exerted by the inflating gas.

Utilizing this concept and construction technique, the degrees of rigidization attainable are broad in range and can be regulated by varying the thickness of the fibrous reinforced resin impregnated member. Since only a small positive pressure differential is required to keep the unrigidized structure expanded, rigidization can be effected in terrestial and other high pressure environments as well as at low pressures or under extreme vacuum conditions.

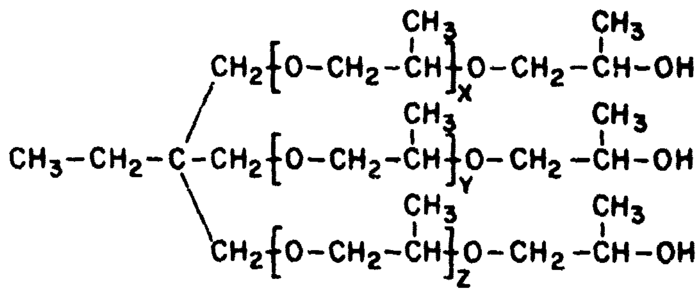
In the initial stages of these programs selected and synthesized polymeric impregnants were evaluated in conjunction with a host of volatile compounds to find appropriate systems for this concept of rigidization. Of the resin intermediates evaluated which included polyesters, polyurethanes, polyepoxies, grafted urethane-epoxy prepolymers and other modified forms of these polymeric intermediates, the polyurethanes and polyepoxies proved to be the most susceptible to the vapor curing process utilizing, respectively, water and triethyl amine as the curing agents. While both of these systems are proposed for the rigidization of expandable structures, the system based on the urethanes is preferred since it can be rigidized at a comparatively fast rate with a nontoxic curing agent. For this reason and because the systems based on the amine-epoxy resin systems were elaborated upon in a previous publication (1), this paper will be limited to discussing the progress made with systems based on the polyurethanes.

Investigation of this concept started with a small internal program. The overall effort of this program was directed at establishing the general feasibility of the proposed concept. Because of the favorable results obtained, the research effort was enlarged under a contract with the Wyandotte Chemicals Corporation. Under these programs systems yielding highly rigid structures were concentrated on because of their broad potential utility. Rigidization mechanisms and synthesis of new impregnants were emphasized rather than the performance of reinforcing fabrics in relation to the rigidizable impregnants or the effects of structures of various geometries.

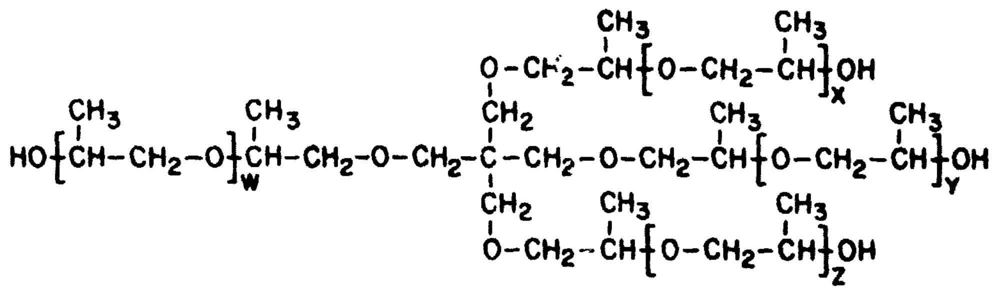
The principal polyurethane resin intermediates evaluated were derived from the polyoxypropylene polyols shown in Figure 1. Aside from the differences in functionality the polyols used to prepare the prepolymers varied in molecular



PROPYLENE GLYCOL BASED DIOL



TRIMETHYLOLPROPANE BASED TRIOL



PENTAERYTHRITOL BASED TETROL

FIG. 1 POLYOXYPROPYLENE POLYOLS

weight. The isocyanate terminated prepolymers were obtained by reacting 2NCO equivalent weights of 2,4 toluene diisocyanate with an OH equivalent weight of the various polyols. This is an addition reaction and is schematically shown in Figure 2.

The basic reason for selecting the polyurethanes as candidate impregnants was their known reactivity with compounds containing active hydrogen. However, as the concept required gaseous curing agents the choice of compounds was automatically limited to water and the more volatile amines. As shown in Figure 3 both types of curing agents lead to urea linkages. Actually the reaction with water is a two step process in which CO_2 is generated. In the first step of this reaction a substituted carbamic acid forms. This acid terminated polymer, in turn, reacts with another isocyanate (NCO) group to form the urea linkage with CO_2 being liberated in the process. As the urea and urethane linkages also contain active hydrogens, these groups under certain conditions will react with NCO units forming (as shown in Figure 4) biuret and allophanate crosslinks. These linkages, however, are not as strong as the urea or urethane type links.

VOLATILE AMINES VERSUS MOISTURE

Semi-quantitative experiments performed early in the program has shown the amines to be unsuitable volatile curing agents for crosslinking isocyanate terminated polyurethane impregnants. The reaction rate with amines was too fast. With the amines thin films of gelled resin quickly formed on the surface which greatly inhibited the diffusion process, consequently the depths of cure obtained after weeks of exposure were very shallow.

Moisture proved to be a very effective curing agent for this type of impregnant. Since the reaction rate with water is comparatively slow, time is allowed for the absorption and diffusion processes to occur before gelation occurs. As a result samples of various polyurethane prepolymers have been observed to cure to a depth of as much as a half an inch.

LAMINATES RIGIDIZED WITH BASIC POLYURETHANE PREPOLYMERS

Impressed by the isocyanate terminated prepolymers' susceptibility to the moisture cure, fibrous reinforced resin impregnants were prepared to obtain basic information regarding the relative cure rates and rigidizing effects of representative members of this family of impregnants. The solvent modified impregnant utilized in the investigation are described in Table 1. The solvent was added to render them easier to handle. Two plies of glass fabric, style 181 were used as reinforcement for the impregnant. Room temperature cures were attempted at 50 percent relative humidity. Only one face of the laminate was exposed to the environment. The other surface was shielded with a moisture impermeable material. To determine the relative rates of cure the flexural properties of the laminates were periodically measured in accordance with ASTM Procedure D790-58T.

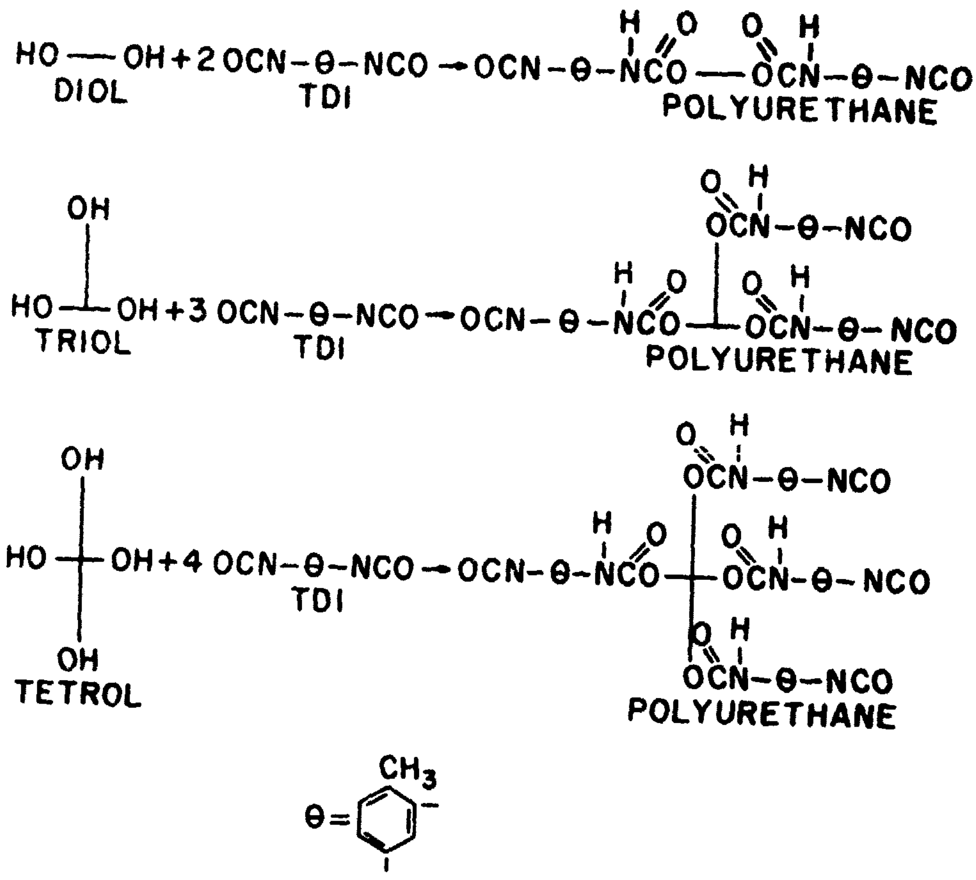


FIG. 2 REACTIONS LEADING TO THE FORMATION OF POLYFUNCTIONAL ISOCYANATE TERMINATED URETHANE PREPOLYMERS.

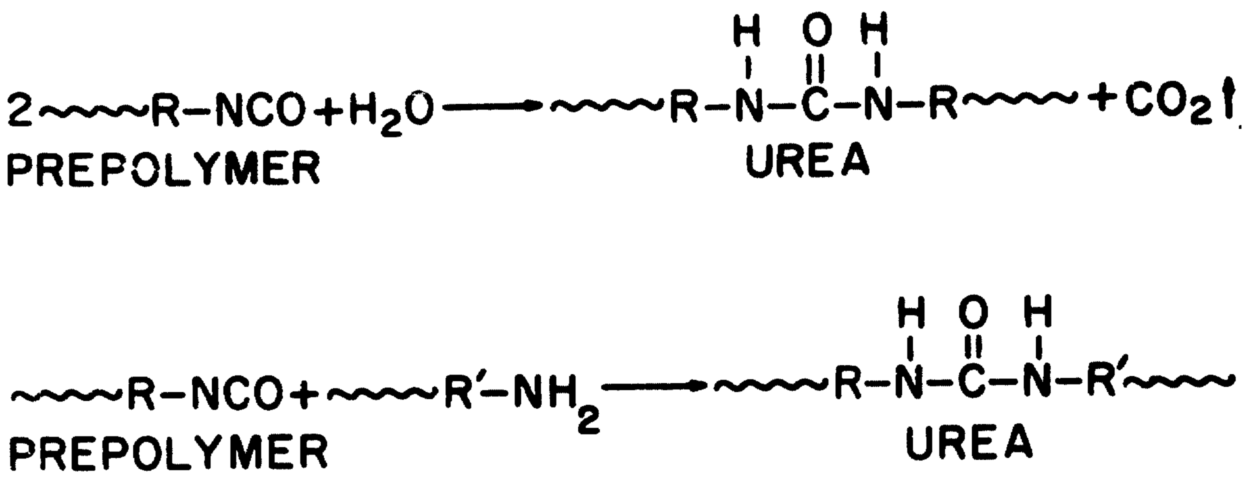


FIG. 3 URETHANE PREPOLYMER REACTIONS WITH WATER AND AMINES

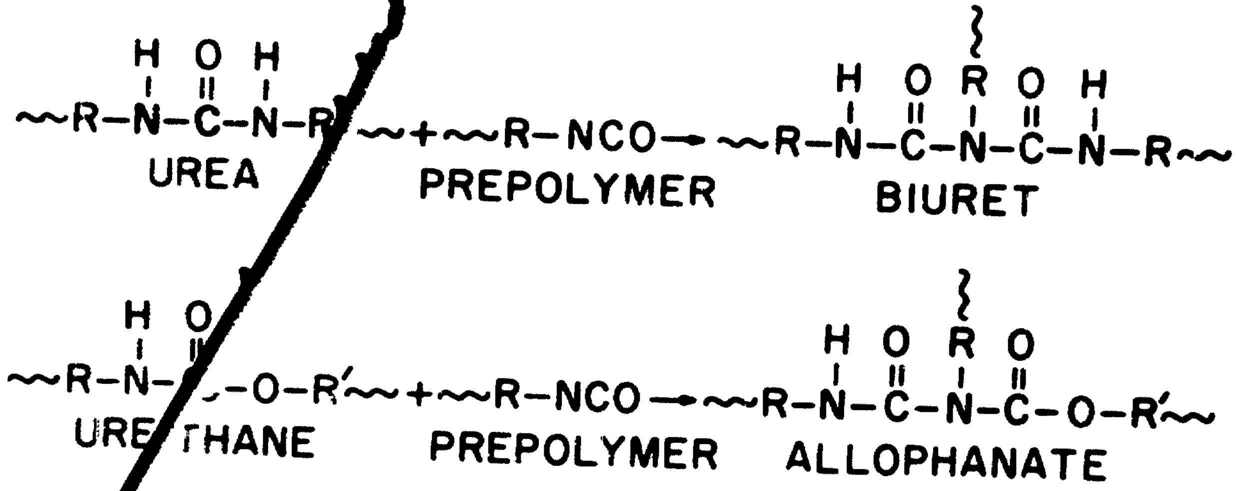


FIG. 4 SIDE REACTIONS OF URETHANE PRE-POLYMERS LEADING TO BIURET AND ALLOPHANATE LINKAGES.

TABLE 1. Composition of Basic Polyurethane Impregnants

Prepolymer	Equivalent Weight/NCO	% Solids	Solvent
TDI Adduct of a 1000 Mol. Wt. Diol	674	90	Toluene
TDI Adduct of a 700 Mol. Wt. Triol	407	90	Toluene
TDI Adduct of a 400 Mol. Wt. Triol	306	80	50/50 Toluene/MEK
TDI Adduct of a 400 Mol. Wt. Tetrol	274	80	50/50 Toluene/MEK

The rigidized strength and the rate of cure of the impregnants increased, in general, with decrease in NCO equivalent weight and increase in functionality (see Table 2). The difunctional high molecular weight impregnant yielded a flexible cured laminate. Resins with a functionality greater than two yielded relatively rigid laminates. These systems, as the fluctuations in flexural data indicate, were lacking in structural integrity due to foaming of the impregnants by the CO_2 liberated in the curing reaction.

TABLE 2. Flexural Properties of Laminates Impregnated with Urethane Prepolymers

Prepolymer	Curing Time, Days	Flexural Strength, psi	Modulus of Elasticity, psi X 10^5
TDI Adduct of a 1000 Mol. Wt. Diol	4	-	-
	7	-	-
	10	480	-
	14	740	-
TDI Adduct of a 700 Mol. Wt. Triol	4	370	-
	7	390	-
	10	1010	-
	14	3590	1.36
TDI Adduct of a 400 Mol. Wt. Triol	4	1840	2.82
	7	2190	0.36
	10	9190	3.99
	14	4600	2.22
TDI Adduct of a 400 Mol. Wt. Tetrol	4	10100	4.23
	7	11350	5.77
	10	8250	3.92
	14	9030	4.18

LAMINATES RIGIDIZED WITH IMPREGNANT BLENDS

Observing that the difunctional high molecular weight impregnant had not foamed, blends of various polyfunctional impregnants were prepared and evaluated in an attempt to obtain nonfoaming systems. A rather large number of blends were evaluated before systems were found in which the rate of the inward diffusion of moisture and the outward diffusion of CO_2 were balanced to yield nonfoamed products. The composition of these nonfoaming impregnant blends are identified in Table 3.

TABLE 3. Composition of Impregnant Blends

	Prepolymer Blends	Mole Ratio	NCO Content, Wt. %
A	NCO Adduct of 700 Mol. Wt. Diol/ NCO Adduct of 400 Mol. Wt. Triol	1/ /2	11.6
B	NCO Adduct of 700 Mol. Wt. Diol/ NCO Adduct of 600 Mol. Wt. Tetrol	1/ /1	10.7
C	NCO Adduct of 700 Mol. Wt. Diol/ NCO Adduct of 700 Mol. Wt. Triol	1/ /2	9.6
D	NCO Adduct of 700 Mol. Wt. Diol/ NCO Adduct of 700 Mol. Wt. Triol	1/ /1	9.3

Data obtained on standard two ply laminates impregnated with these systems are presented in Figure 5. While it was not absolutely necessary these prepolymer blends were diluted to 80 percent solids to render them easier to handle using dried toluene as the solvent.

The properties of these laminates were far superior to those of the previous laminates impregnated with the systems which foamed. Laminates impregnated with the blends attained optimum cured strength within approximately 7 days. The reactive cure rate and the ultimate state of rigidity attained increased as the NCO/equivalent weight percent of the blends increased. The most effective blend yielded a laminate which attained approximately 50 percent of the strength of laminates conventionally prepared utilizing the best of laminating plastics available today.

CATALYZED URETHANE IMPREGNANT BLEND

While the rates of rigidization of the above systems would be satisfactory under many circumstances, consideration of failures in a space environment resulting from early micrometeoroid punctures, prompted the evaluation of systems

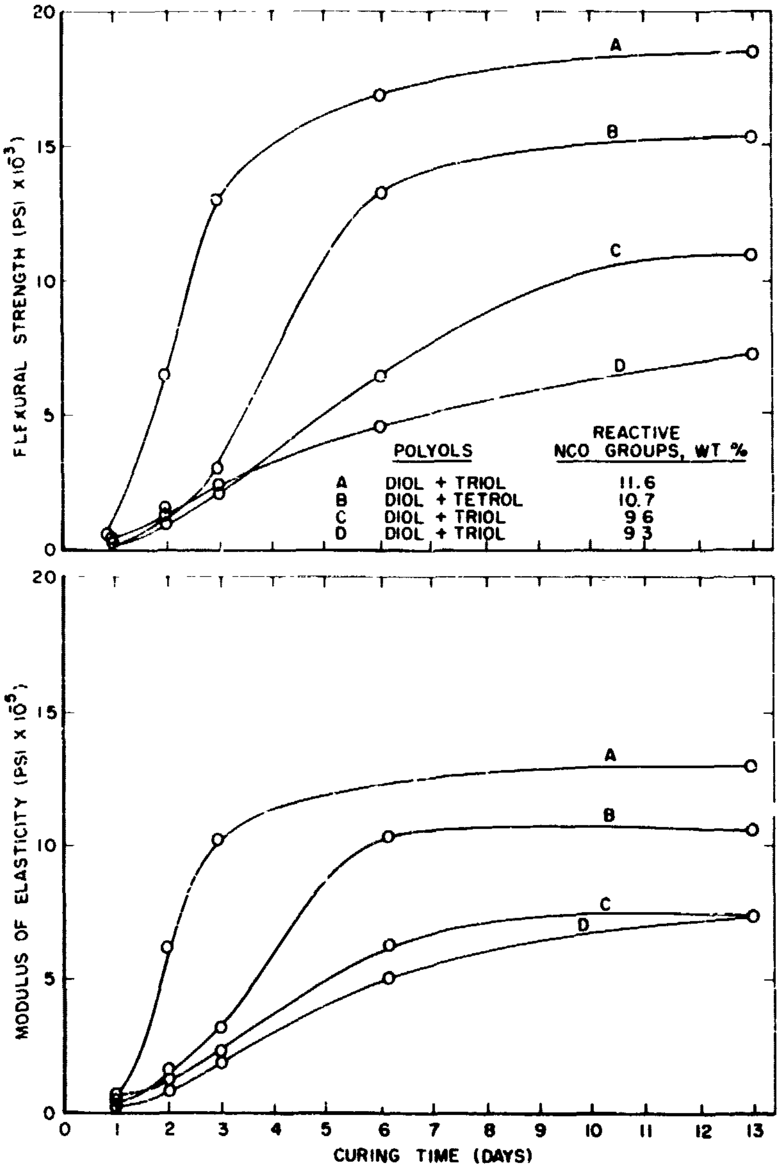


FIG. 5 FLEXURAL PROPERTIES OF LAMINATES IMPREGNATED WITH URETHANE PREPOLYMER BLENDS

modified with compounds, which experience has indicated catalyze the H_2O urethane reaction.

The fastest curing prepolymer blend which yielded the most rigid laminate was utilized in the evaluation of the selected catalyst compounds. The compounds evaluated were: 2,2,2,-diazabicyclooctane, dibutyltin dilaurate and 1,2,4-trimethylpiperazine. Flexural property data obtained on laminates impregnated with samples of this prepolymer blend containing 0.5 percent by weight of the candidate compounds are shown in Figure 6.

All three compounds accelerated the rigidization reaction. Of the compounds evaluated, although not the most effective, trimethylpiperazine was the preferred catalyzing agent. Dibutyltin dilaurate catalyzed systems, the results from other work has shown, yield cured products which are, relatively speaking, thermally unstable. While diazabicyclooctane was the most effective catalyst, as will be evident later, the system containing this modifier had a comparatively short pot life.

Laminates impregnated with the catalyzed systems attained the greatest portion of their cured strength within approximately 24 hours. As the flexural data show, the ultimate degree of rigidity attained by laminates impregnated with the catlayzed systems was somewhat lower than that attained using the uncatalyzed resin blend as the impregnant. The comparatively low strength of the catalyzed systems is attributed to catalyst induced secondary chemical reactions which result in the formation of the relatively weak biuret and allophanate type linkages.

MOISTURE PERMEABLE CANDIDATE INNER LINERS

With the attainment of promising moisture curable impregnants, experiments were initiated to find suitable inner barrier materials. So as not to delay the rigidization process it is essential that this structural component be highly permeable to the vapor curing agent. As mentioned earlier, it is employed primarily to keep the tacky inner wall surface of the resin impregnated segment of the structure from touching and adhering in the packaged state. By transmitting the pressure exerted by the inflating agent it also forces the plies of the structure together on inflation.

The relative permeabilities of the selected candidate materials are given in Table 4. These measurements were made at room temperature using the wet cup method. Polyethylene oxide film was by far the most permeable to moisture. Theoretically this material is more than adequate for this use. Assuming that a structure contains 2 ounces of the most promising candidate impregnant per unit area, the stoicheometric amount of moisture required for complete reaction would penetrate the polyethylene oxide barrier film in less than an hour.

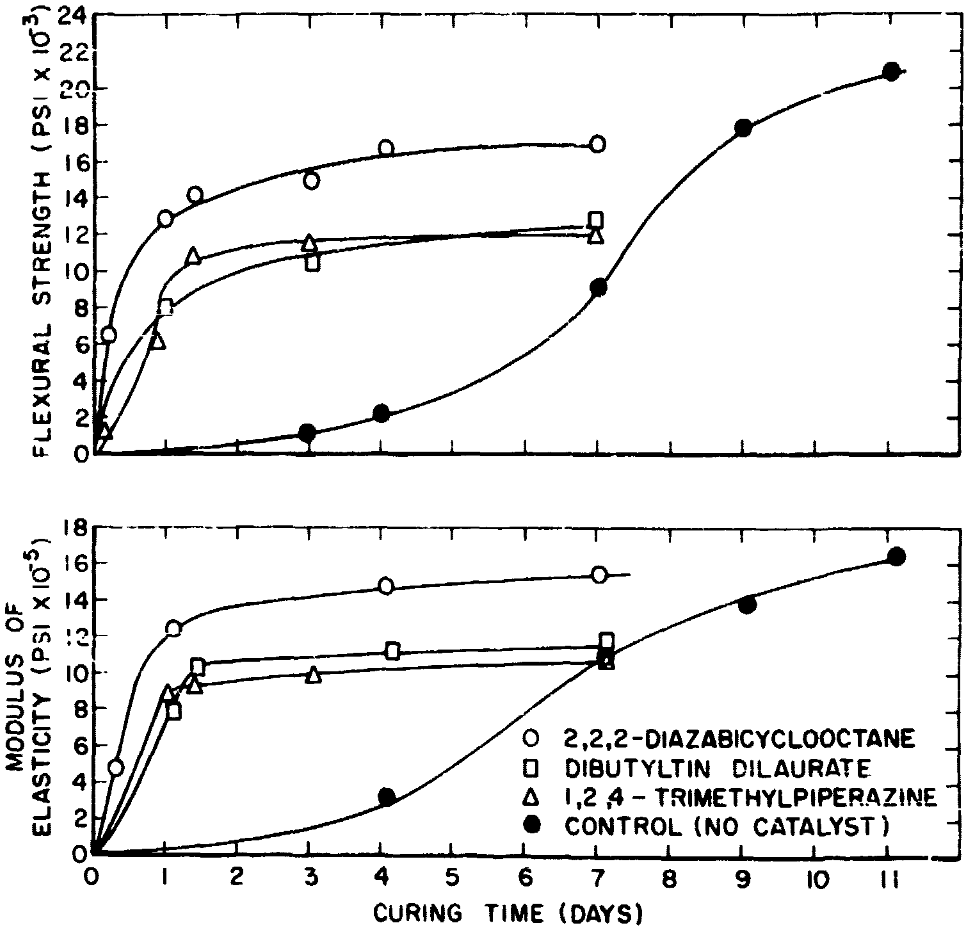


FIG. 6 FLEXURAL PROPERTIES OF LAMINATES IMPREGNATED WITH A CATALYZED URETHANE PREPOLYMER

TABLE 4. Moisture Permeability of Candidate Inner Barrier Materials

Inner Barrier Materials	Permeability*
Polyethylene Oxide Film	15.07
Cellulose Acetate Film	4.98
Methocel Film	4.66
Ethyl Cellulose Film	1.83
Polyethylene Oxide Coated Nylon Fabric	7.36

* $gm/hr/ft^2/mil$ thickness

STABILITY OF IMPREGNANTS IN STORAGE

From the beginning of the various experiments samples of the candidate impregnants were packaged in glass containers and stored under anhydrous conditions so that their relative stabilities could be examined over a substantially long period.

Some of the more promising systems were stored for periods as much as six months. On the basis of viscosity measurements, the uncatalyzed prepolymeric blends are stable for periods in excess of six months. Storage stability of the catalyzed systems are shown in Table 5. The systems catalyzed with 1,2,4-trimethylpiperazine and dibutyltin dilaurate proves to be adequately stable. These systems changed very little in viscosity during the five month storage period.

TABLE 5. Effects of Catalyst on the Stability of a Urethane Prepolymer Blend

Catalyst	Gel Time (75°F)
Lead Naphthenate	2 hours
2,2,2,-Diazabicyclooctane	7 days
Stannous Octoate	6 days
1,2,4-Trimethylpiperazine	5 months
Dibutyltin Dilaurate	5 months

BARRIER SHIELDED RIGIDIZABLE ASSEMBLIES

Additional data regarding the performance of the candidate inner barriers were obtained by using polyethylene oxide and methocel films as shields for two ply glass fabric laminates impregnated with the most effective catalyzed and

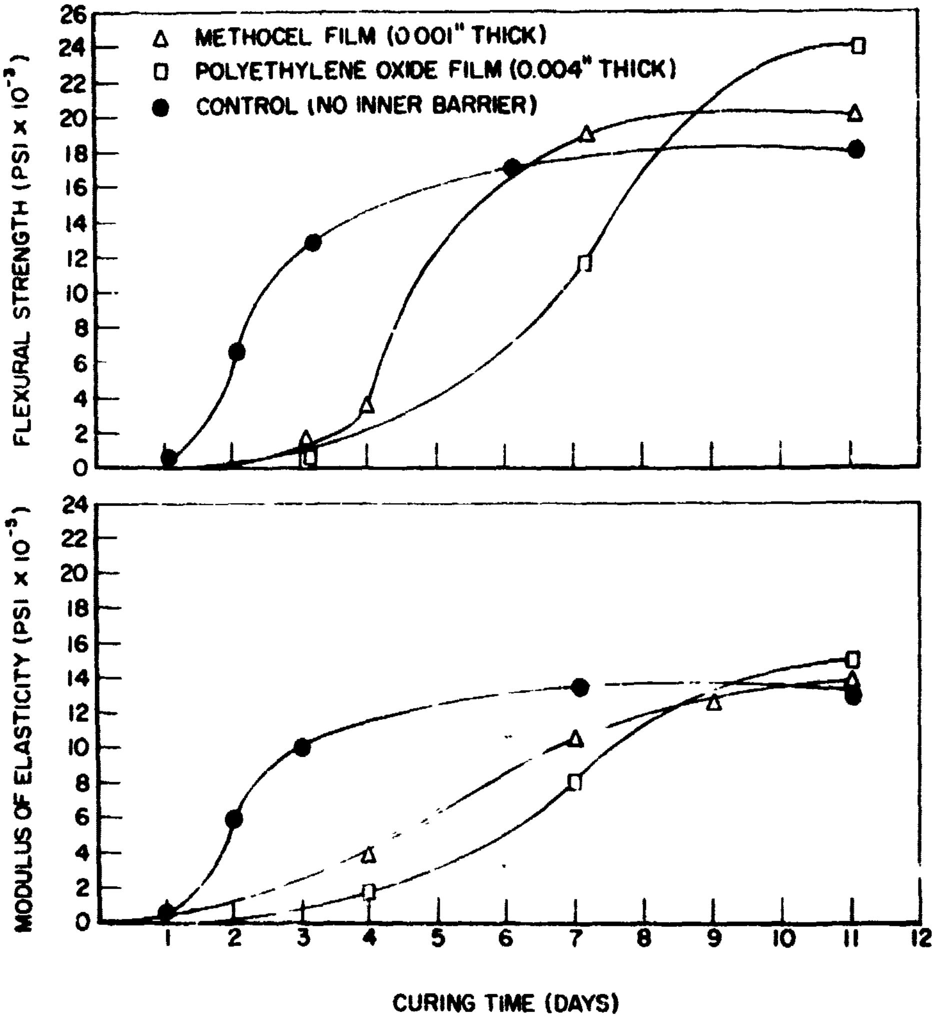


FIG. 7 FLEXURAL PROPERTIES OF FILM SHIELDED LAMINATES IMPREGNATED WITH AN UNCATALYZED URETHANE PREPOLYMER

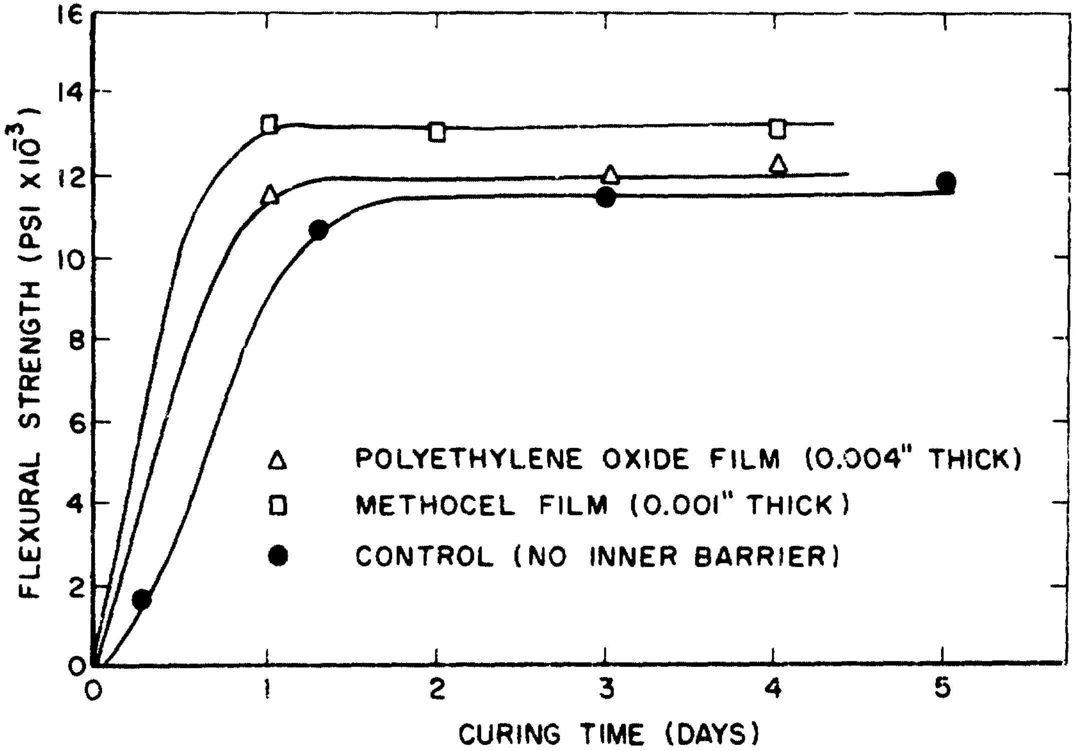


FIG. 8 FLEXURAL STRENGTH OF FILM SHIELDED LAMINATES IMPREGNATED WITH A CATALYZED URETHANE PREPOLYMER

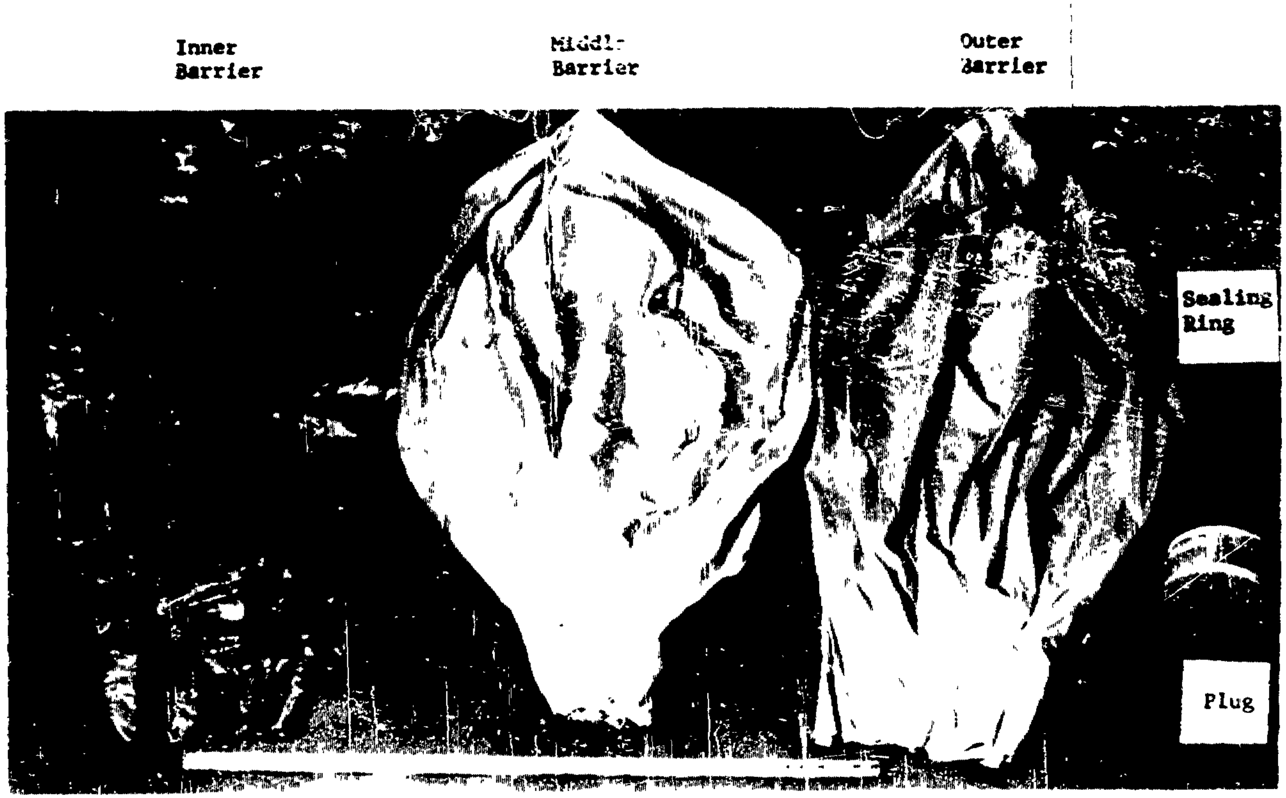


FIG. 9 COMPONENTS OF RIGIDIZABLE STRUCTURE

uncatalyzed prepolymer systems. The laminates prepared were conditioned and evaluated in the manner previously described.

The barrier films prolonged the cure of the uncatalyzed impregnant 2 to 3 days. The cure rate of the catalyzed impregnant was slightly enhanced by the shielding materials. As shown in Figures 7 and 8 the ultimate flexural properties of the shielded laminates were somewhat better than those of the unshielded control sample. Based on the overall results either of these materials would be satisfactory for use as an inner barrier for expandable structures impregnated with moisture curable polyurethane impregnants, especially in conjunction with the catalyzed systems.

RIGIDIZATION IN A SIMULATED SPACE ENVIRONMENT

To obtain more realistic information regarding the performance of this type of rigidizing agent, a three foot diameter spherical shaped model structure was assembled and subjected to rigidize under simulated vacuum conditions at room temperature. The components of this structure are shown in Figure 9. Polyethylene oxide film was used as the permeable inner liner. A single ply of glass fabric, style 184, was utilized as reinforcement for the impregnant and a Hypalon coated nylon fabric served as the impermeable cover for the structure. A prepolymer blend (blend A, Table 3) catalyzed with a 0.5 percent by weight of 2,4,6-trimethylpiperazine was utilized as the rigidizable impregnant. The sleeve-like opening of the assembled structure was sealed with the plug fitted with the inlet valve shown in the photograph. It was held in position by the adjustable ring clamp.

The structure and the altitude chamber were simultaneously exhausted of air to pressures respectively of 2.6 and 10 mm of Hg. Water vapor was then introduced to completely inflate the structure. The ultimate pressure attained was 44 mm of Hg. These conditions were maintained essentially throughout the experiment by adding moisture saturated air to compensate for losses due to a slight leak in the structure.

Upon removal after a 16 hour exposure period the structure Figure 10 was very rigid, which proves the suitability of this type of rigidizing system and the practicality of the overall concept.

FUTURE EXPERIMENTAL WORK.

Investigations into the rigidization process involving the moisture curable urethane impregnants is being continued. Experiments are currently being conducted to determine the effect of temperature on the rate of rigidization and the strength of fibrous reinforced structural assemblies. Techniques for preparing one component rigidizable structural assemblies utilizing an integrally woven honeycomb fabric coated on both sides with a flexible impermeable polymer are being investigated. The honeycomb type of construction has the advantage, of course, of high compression resistance.

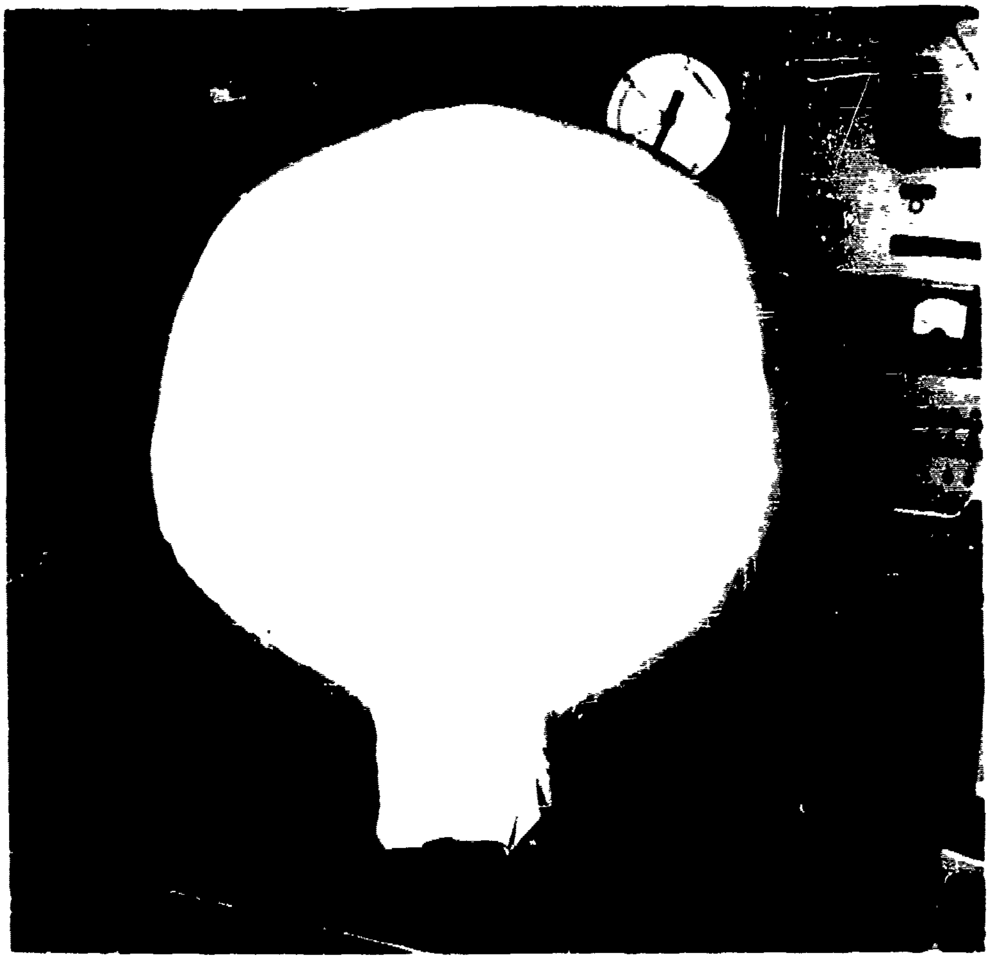


FIG. 10 RIDIGIZED STRUCTURE

REFERENCES

1. "Rigidizable Expandable Aerospace Structures," T. L. Graham and R. G. Spain, Society of Automotive Engineers, Inc., National Aerospace Engineering and Manufacturing Meeting, 8-12 October 1962.

EXPANDABLE LUNAR SHELTER CONCEPTS

A Collaborative Design Research Project by the Department of Architecture and the Department of Industrial Design of the College of Design, Architecture, and Art of the University of Cincinnati with the cooperation and assistance of the Flight Accessories Laboratory, A.S.D., U.S.A.F.

Karl H. Merkel, Professor of Architecture
James M. Alexander, Professor of Industrial Design

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Personnel of the Flight Accessories Laboratory, A.S.D., U.S.A.F., met with officials and faculty of the College of Design, Architecture, and Art of the University of Cincinnati in 1962 to determine the feasibility of an unfunded student design study of lunar shelters.

The college recognized the educational advantages offered by the project and decided to use it as a learning device to stimulate the creative efforts of the students. It presented an opportunity to familiarize the students with some of the problems the space scientists and engineers are facing and to encourage them toward research and applications of architectural knowledge in the space program. The broad aim of education for both architectural and industrial design students is to supply the training, through the arts and sciences, so they may design satisfying physical, psychological, and cultural environments and facilities for human existence. To help achieve this aim, the student is exposed (1) to studies concerning knowledge of structures, materials, and mechanisms; (2) to studies of the physical and psychological effects of spaces, shapes, textures, colors, climate and environment, etc.; (3) to experiences which help develop imaginative, creative, and intuitive design abilities; (4) to a liberal education in the Humanities and citizenship; and (5) to situations and experiences which require his understanding of and ability to organize the inter-relationship of the preceding studies.

The students exposed to this design study of Lunar Shelters were twenty-nine fourth year architectural and twenty-four third and fourth year industrial design students, all enrolled in co-operative courses at the college. The fifty-three students were grouped in thirteen teams, and were assigned the Program* which was developed by the Flight Accessories Laboratory, Aeromedical Laboratory, and Flight Dynamics Laboratory.

The program stated the purpose of the project was to study methods of providing shelter for the personnel, materiel, and supplies for a manned lunar base. The shelter was to accommodate a crew of nine men with all necessary equipment and provisions for a thirty day exploratory period on the moon.

*Lunar Shelter Concepts Univ. of Cin., F.A.L. A.S.D. 1962 pp. 4 - 8

In order to provide as much time as possible for actual shelter design research, the program described certain developments and technical requirements that the students had to meet and certain assumptions to use as criteria. Also the following five suggested materials and/or structural techniques were offered for the students to investigate and perhaps to use or to propose further developments:

1) Prefabricated modular units, particularly those of cylindrical form that might be compatable with the rocket shape. Investigation of and suggestions for the erection or assembly of these units under lunar environment were required of the students.
2) Goodyear Aircraft Company's "Airmat" inflatable double-walled baloons that would be lightweight and easily erected Proposals were required for eliminating the possibility of structural failures due to punctures.
3) Polyurethane foamed-in-place structures with proposals for keeping the volume and weight of the material to a minimum.
4) Expandable honeycombed structures with proposals for keeping the weight and erection techniques to a minimum.
5) Unfurlable or telescoping units with proposals for keeping the weight and erection mechanics and techniques to a minimum.

In order to familiarize the students with some of the techniques, equipment, and other technical information, the Flight Accessories Laboratory personnel visited the college several times for conferences, provided literature, and arranged meetings for the students to consult with personnel of other space program agencies and laboratories. Further research by the students was done at the University Observatory and Department of Astronomy, and other colleges on the campus.

The design study was scheduled for a seven week Co-operative school term. The time allotted for the study was relatively short but the good results indicated that architects and industrial designers would be capable of making tremendous contributions toward the planning and designing necessary to satisfy the physical, psychological, and cultural environments and facilities for human existence under almost any set of conditions.

At the end of the seven week period the students presented their team solutions in the form of drawings, models, photographs, and color slides, together with a paper describing the design concept philosophy, the packaging and erection techniques, the materials and structures, the environmental protection, the interior planning and furnishings, the integration of equipment, and other statistical data. Figures 5 through 31 of the Appendix together with accompanying comments illustrate and describe four of the thirteen solutions submitted.

The program requirements called for the packaged shelter, provisions and equipment to fit into a cylinder shape not to exceed fourteen feet in diameter by thirty-five feet in length (5390 cu. ft.) and that the weight of these items would not exceed twenty-five thousand pounds. The crew would arrive along with a lunar traversing vehicle in separate rockets. The volume and weight limitations were used to the maximum and the packaged units had practically no voids. Three of the thirteen solutions were

able to develop over twelve hundred square feet of sheltered area from this package. One solution proposed under six hundred square feet as being sufficient for the needs. The greatest volume of sheltered space was over eleven thousand cubic feet. The average proposed was about nine hundred square feet, and the average volume of the shelter was about eight thousand cubic feet.

The biggest ratio of packaged volume to expanded volume was 1 to 2, with an average of about 1 to 1-1/2. As this ratio increased, the need for more air and more equipment increased, thus reducing the amount of space in the carrier for the packaged shelter materials. As the ratio decreased, the volume assigned to the packaged shelter could be increased. A balance of space required for the packaged equipment and supplies and the space required for the packaged shelter materials had to be determined to permit the desired area and volume for the crew. The average area per man was one hundred square feet, and the average volume per man was nine hundred cubic feet, including the space for equipment and supplies. The open space for human use averaged better than 60% of the total shelter. This perhaps might be termed luxurious when considering the requirement for providing a totally controlled atmospheric environment, however physical and psychological needs for the nine men involved in this long a stay in such a possibly nerve-shattering situation require more than bare minimum space and facilities.

Eight of the solutions developed the most obvious approach of using the rocket cylinder itself as the shelter. The assigned cylinder size was considered by most teams to be not adequate in volume or area to accommodate the equipment plus the crew, therefore some alterations of the cylinder would be necessary upon arrival on the moon. Variations of the cylinder shelter included methods of expanding the cylinder by telescoping the ends (figure 1), projecting elements from the sides of the cylinder (figure 2), splitting the cylinder longitudinally and expanding the halves (figure 3), and dividing the cylinder transversely into horizontal units and connecting the units with tunnels (figure 4).

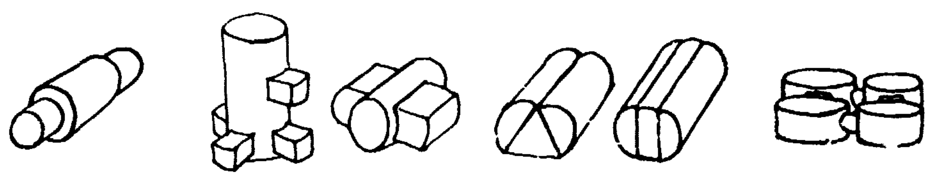


fig.1. fig.2 fig.3 fig.4

Variations of Cylinder Type Shelters

One major problem that evoked several ingenious solutions for the designs using the cylinder on its side was that of manoeuvering the cylinder from its vertical landing position to a horizontal position to eliminate the use of inside ladders for circulation of the crew. This was accomplished by using cranes or hoists incorporated in the cylinder or by using the lunar traversing vehicle as a manipulator. A few suggestions for operation of lifts or elevators, as well as for erection of some inflatable structures, considered the differential of internal and external air pressures as a means of providing power.

Methods of protecting the structure and occupants from radiation, injury, or damage varied from burying the entire structure to using above-surface units with "storm cellars" or to using stockaded and umbrella shielded devices around and over the shelter units.

Several solutions involved unfurlable or inflated and foamed-in-place structures separated from the rocket cylinder. Most of the solutions made use of combinations of two or more of the various currently marketed or researched materials or ideas, applying them where they fit specific needs best. The university and the students were not obligated nor influenced to use any particular makes of materials or structures, therefore they were in a position to make unbiased evaluations of the adaptability of available materials for specific applications. This led to a good variety of solutions and to flexibility in combining materials and techniques. Although some of the solutions had certain similarities (mainly due to the use of similar materials or structural theories previously mentioned) they were all distinctly different approaches to the problem, particularly in the arrangements or relationships of the interior spaces and in the equipment and furnishings.

Among the factors determining the solutions were several that most teams considered very important although they were not set forth as requirements. These included;

1) The structure must arrive on the moon in as nearly complete a stage as possible and that any erection techniques necessary should be simple and automatic or remotely controlled using a minimum of moving parts.
2) During transit any equipment in the shelter should be in its final or almost final position for use upon arrival on the moon and that it should be arranged in the package or carrier cylinder to achieve balance of weights to not cause the rocket in flight to roll or topple.
3) The package or carrier cylinder should be so arranged that when equipment must be relocated for use on arrival and erection, the resulting void becomes working-living space for the crew.
4) The energy or power supply should be located in such positions as to become immediately active and available on arrival for use in erection of the shelter.
5) Exhaustable items (i.e. food, air, water, clothing, etc.) should be stored or located as to be easily removed and replaced if the shelter were to be used by subsequent crews or for prolonged stays.
6) The shelter should be equipped with emergency exit devices for evacuation of the crew in case of damage to the shelter.
7) Monotony, homesickness, lethargy, etc., should be avoided or counteracted by use of both familiar and unfamiliar interior space shapes and sizes, and by judicious use of color and textures of materials, furniture and equipment inside the shelter.
8) Arrangement of interior spaces should primarily provide for efficient work operations but should be so designed as to permit the crewmen to vary their immediate environment and activities.

One of the most conspicuous notes about all the projects was the concern for the human being in terms of some of the indefinable qualities of aesthetics and of experiencing the psychological as well as physical aspects of his surroundings. These particular concerns may well be the ones where the architect and industrial designer may be best qualified to contribute toward the space conquest.

The variety of solutions presented in the study seems to indicate that persons trained in the "design fields" are capable of producing fresh approaches and "ideas" to the already established facts that lunar shelters are a technical feasibility. The solutions in the study are definitely only "concepts" based on scanty knowledge of the moon's environmental conditions and on the use of materials and structural systems that must still be tested under such conditions or perhaps must still be invented. Since the time period for investigation is short, if 1970 is the date set for establishing a lunar base, Lunar Shelter "Concepts" or "ideas" must be proposed, investigated, and developed directly and in haste, considering proposals from every possible source.

The thirteen solutions have been collected, edited, and published by the Flight Accessories Laboratory in a volume entitled "Lunar Shelter Concepts", and are submitted as ideas which might be engineered and developed into practical, workable products.

FIGURE 5

Designers:
D. Malone, Arch.'64
J. Spinnenweber, Arch. '64
H. Fischer, Ind. Des. '63
D. Moehring, Ind. Des. '64

Solution of Team #2

DESIGN PHILOSOPHY

The concept of this team is based on four basic factors: first, the structure should arrive on the moon in as fully a completed stage as possible; second, whatever erection techniques required on the lunar surface should be as simple as possible; third, the shelter as an entity should function as a precision machine with a minimum of human attendance; fourth, the structural capabilities of the transporting cylinder should be utilized to their fullest.

Therefore, this concept involves a basically completed shelter before it actually lands on the moon. The mechanics of transforming the cylinder into working and living space are simple and can be accomplished in a "shirt-sleeve" environment.

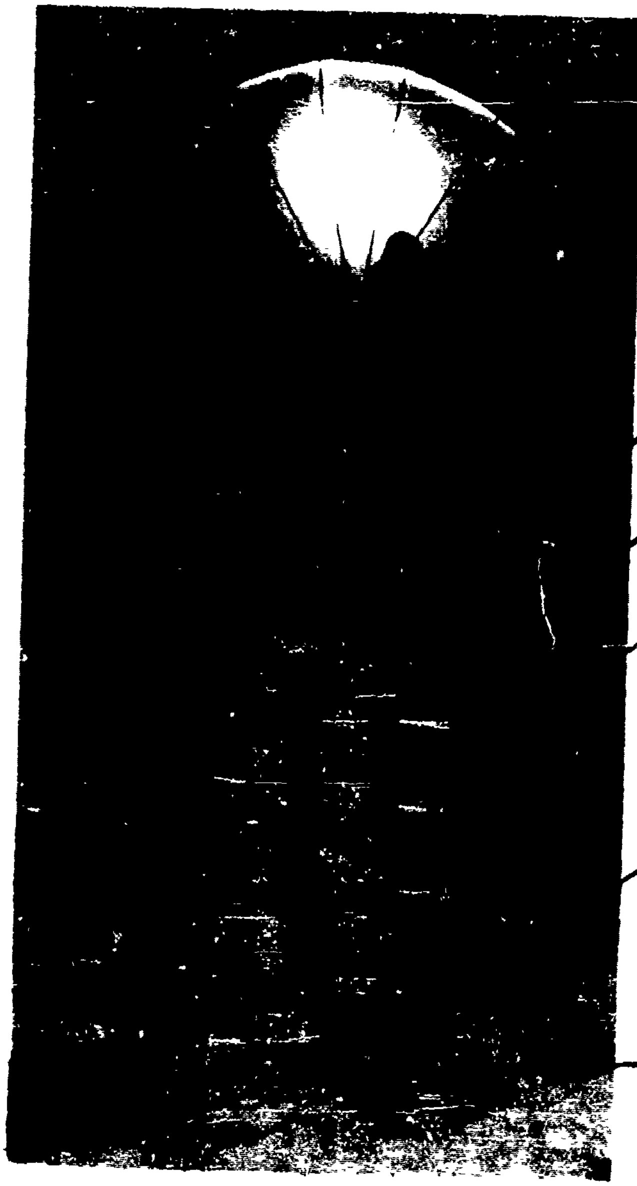

Segmented parabolic solar reflector in open position on mast elevated from stem of shelter cylinder.
Observation domes
Typical extended "drawer unit" (pod)
35' long x 14' diam. rocket cylinder
Typical extended airlock and stabilizer. In installation would be buried under 10' lunar dust to form protected "storm cellar" in event of solar flares or meteorite bombardment
Multi-purpose vehicle for scooping lunar dust, placing "black body" thermal radiation absorbers, etc.

FIGURE 6

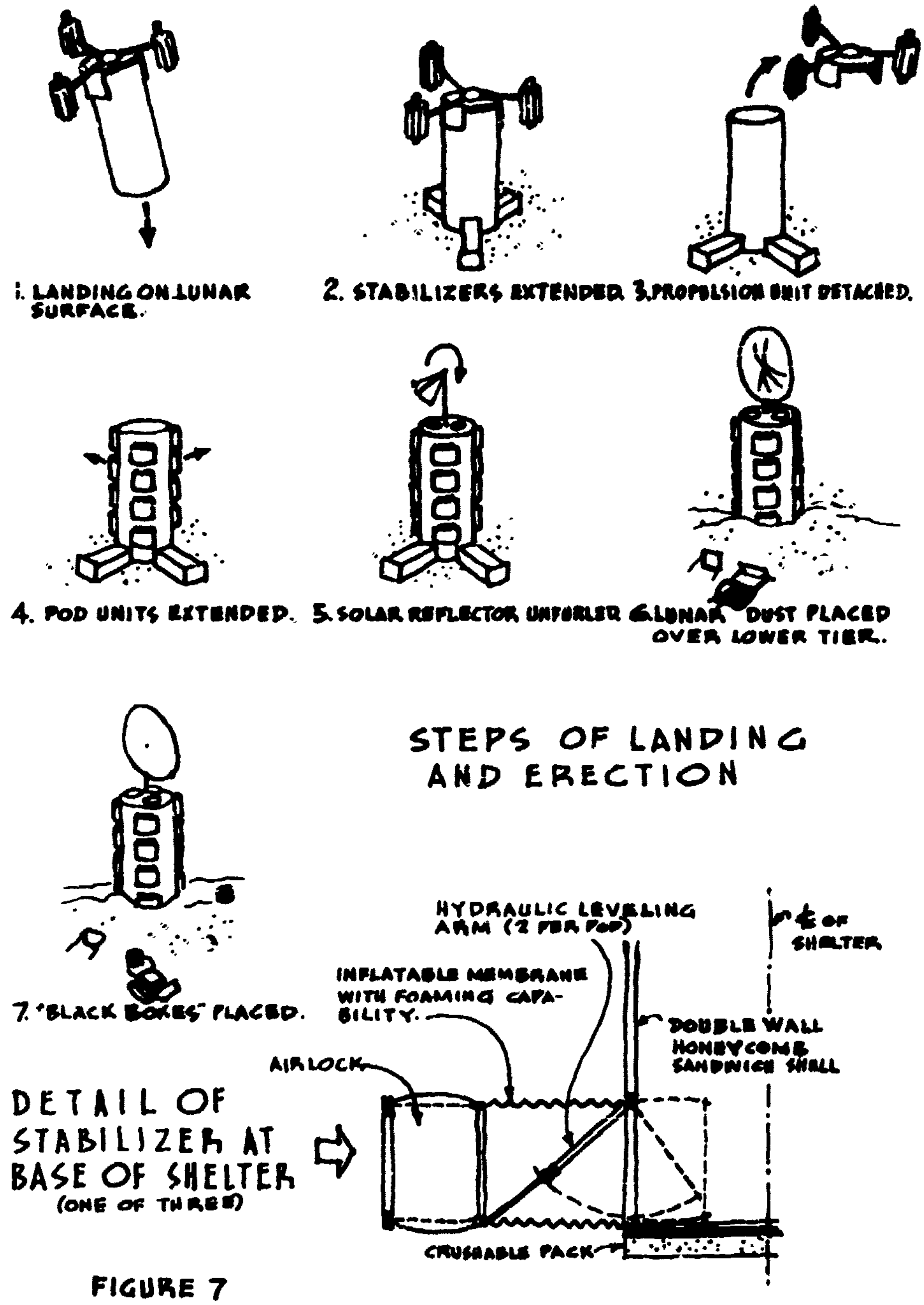

1. LANDING ON LUNAR SURFACE.
2. STABILIZERS EXTENDED.
3. PROPULSION UNIT DETACHED.
4. POD UNITS EXTENDED.
5. SOLAR REFLECTOR UNFURLED
6. LUNAR DUST PLACED OVER LOWER TIER.
STEPS OF LANDING AND ERECTION
7. 'BLACK BOXES' PLACED.
HYDRAULIC LEVELING ARM (2 PER POD)
℄ OF SHELTER
INFLATABLE MEMBRANE WITH FOAMING CAPA-BILITY.
DOUBLE WALL HONEYCOMB SANDWICH SHELL
AIRLOCK
DETAIL OF STABILIZER AT BASE OF SHELTER (ONE OF THREE)
CRUSHABLE PACK

FIGURE 7

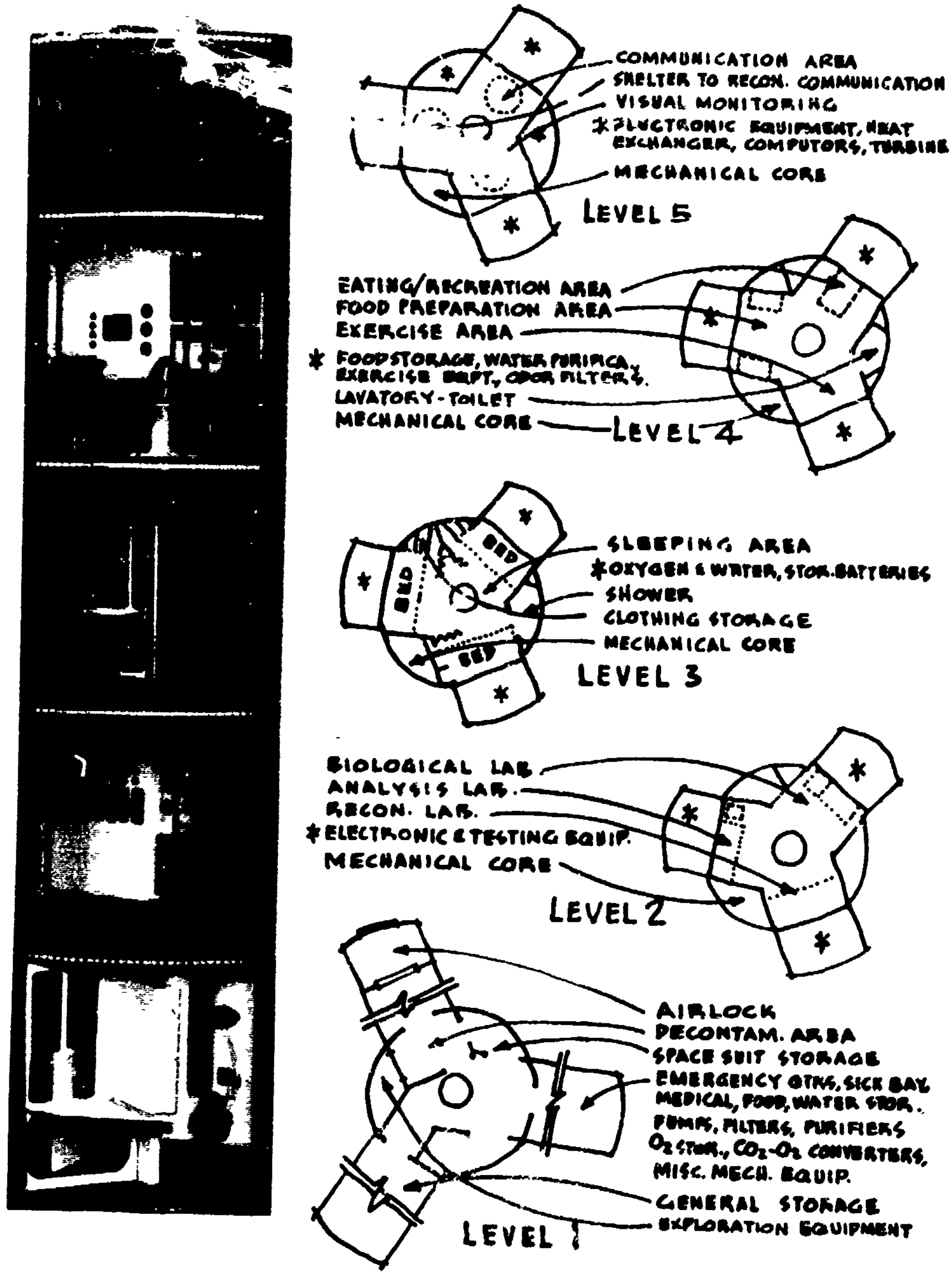

COMMUNICATION AREA
SHELTER TO RECON. COMMUNICATION
VISUAL MONITORING
* ELECTRONIC EQUIPMENT, HEAT EXCHANGER, COMPUTORS, TURBINE
MECHANICAL CORE
LEVEL 5
EATING/RECREATION AREA
FOOD PREPARATION AREA
EXERCISE AREA
* FOODSTORAGE, WATER PURIFICA., EXERCISE EQPT., ODOR FILTERS.
LAVATORY-TOILET
MECHANICAL CORE
LEVEL 4
SLEEPING AREA
* OXYGEN & WATER, STOR. BATTERIES
SHOWER
CLOTHING STORAGE
MECHANICAL CORE
BED
LEVEL 3
BIOLOGICAL LAB.
ANALYSIS LAB.
RECON. LAB.
* ELECTRONIC & TESTING EQUIP.
MECHANICAL CORE
LEVEL 2
AIRLOCK
DECONTAM. AREA
SPACE SUIT STORAGE
EMERGENCY QTRS, SICK BAY, MEDICAL, FOOD, WATER STOR.
PUMPS, FILTERS, PURIFIERS
O_2 STOR., CO_2-O_2 CONVERTERS,
MISC. MECH. EQUIP.
GENERAL STORAGE
EXPLORATION EQUIPMENT
LEVEL 1

FIGURE 8

Designers:
L. Fabbro, Ind. Des. '64
R. Kreinbrink, Arch. '64
J. Madzula, Arch. '64
P. Thornton, Ind. Des. '64

PACKAGE AND ERECTION TECHNIQUES

In the packaged state each unit is supported spearately. This enables the ship to control the position of its payload; also it insures that no one unit supports any more than its own weight under acceleration.

The method of packaging the ship consists of sliding each unit down vertical tracks to a position where it will be locked automatically and be supported individually. Upon the completion of packing, the nose cone is set in place. The unpacking process is the reverse by use of the traversing vehicle.

Erection of the units shall take place near the perimeter of a small crater. The site shall be graded level by the vehicle and the air lift hole shall be excavated by the vehicle before the air lock unit is positioned. The first unit to be positioned is the air lock unit with the mechanical and unit connectors hinged at its base. When removed from the ship, the units shall automatically expand vertically and lock into position. The remaining four units are positioned and connected to the air lock and each other by the mechanical and unit connectors. The air lock is the first unit to be expanded horizontally. This enables the removal of the expandable solar collector, generator, and white and black bodies, for their erection. The prime interest is the operation of the mechanical equipment for the function of all units before their final expansion.

Inflation of the air mat structure follows the operation of mechanical equipment. This enables correct alignment and check for air leaks in all units. The air mat and rigid panels are then rigidized from within by actuation of pellets of expandable foam.

Upon rigidizing of all units, a structure of stabilized lunar material shall be poured over entire complex by the traversing vehicle. The vehicle shall be capable of stabilizing the lunar material with foam and ejecting this material to a depth sufficient to aid the support of the remaining cover material. The total covering shall consist of approximately 2 feet of stabilized lunar material and 8 feet of existing lunar material.

During the covering process, part of the team will be in the complex placing floor sections over the expanded area, positioning equipment, wall section, and furniture. This erection process can be handled in stages to enable work and eat/sleep cycles to begin.

STRUCTURES AND MATERIALS

The main structural component for each unit shall be a central core or wall between rigid floor and ceiling. Additional structure shall be provided by a layer of stabilized lunar material.

The structural materials are broken into two main categories. First is the rigid material which is molded and fabricated of a high strength, light weight material. Second, the expandable material will consist of an integration of air mat and expandable foam reactants inside the air mat.

We feel that this combination achieves the rigid and light weight properties needed in this structure.

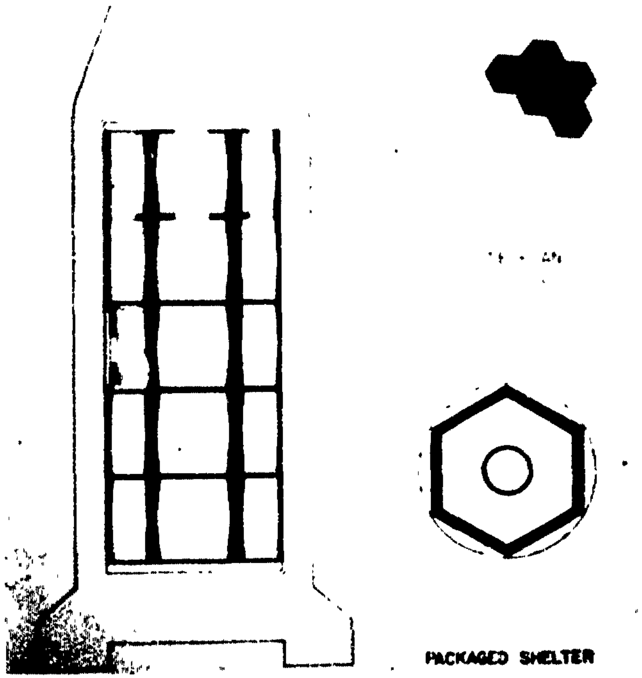


FIGURE 9

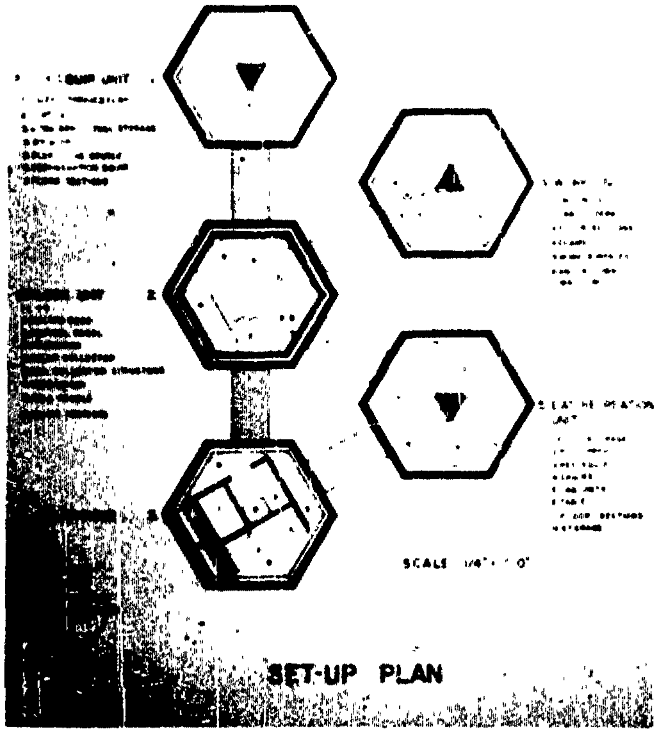


FIGURE 10

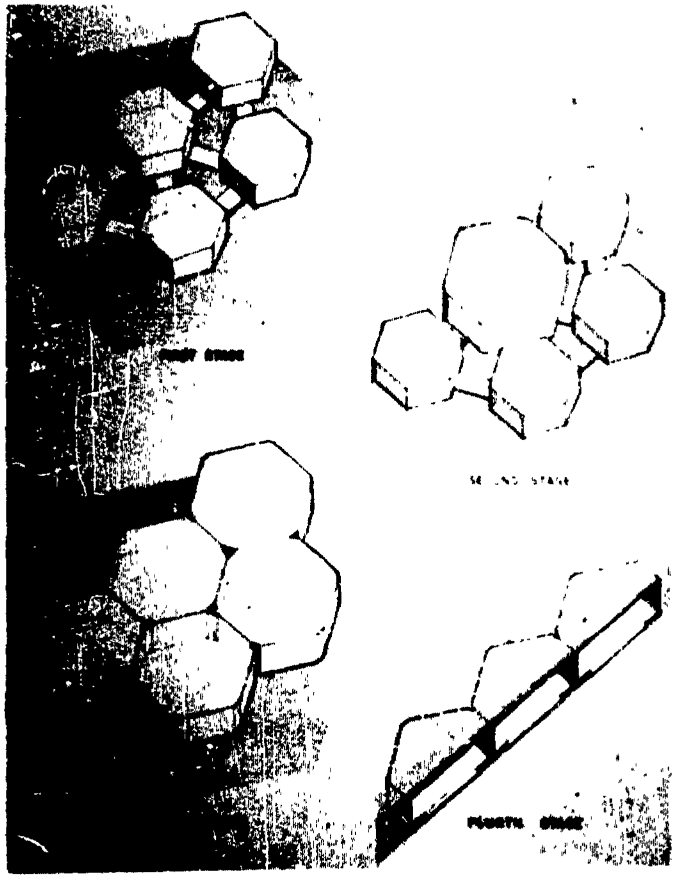


FIGURE 11

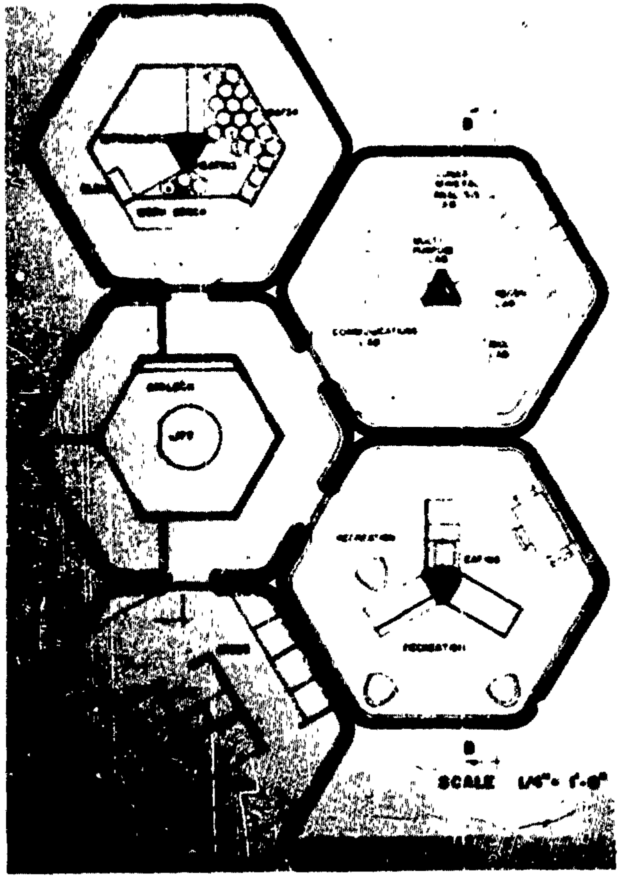


FIGURE 12

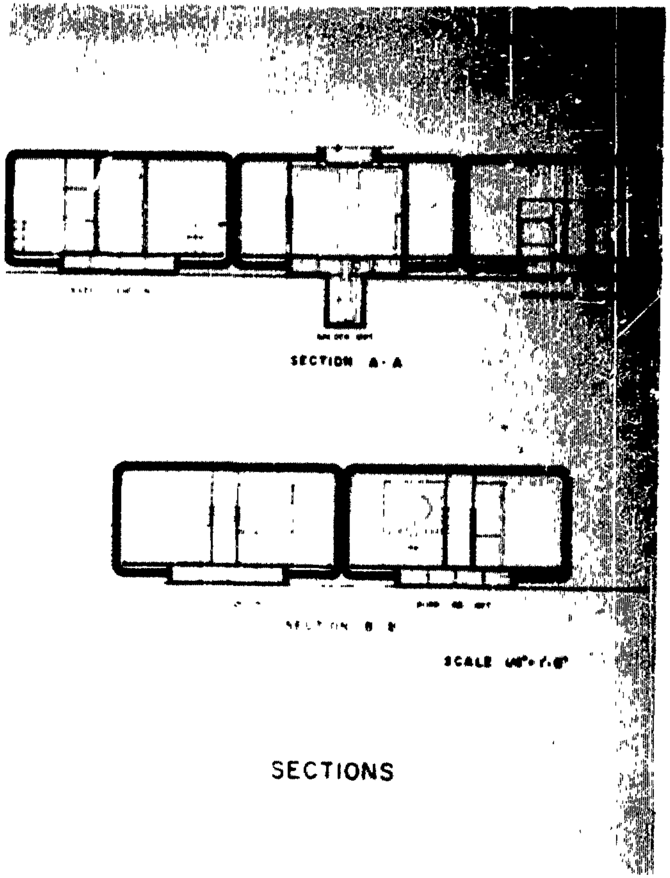


FIGURE 13

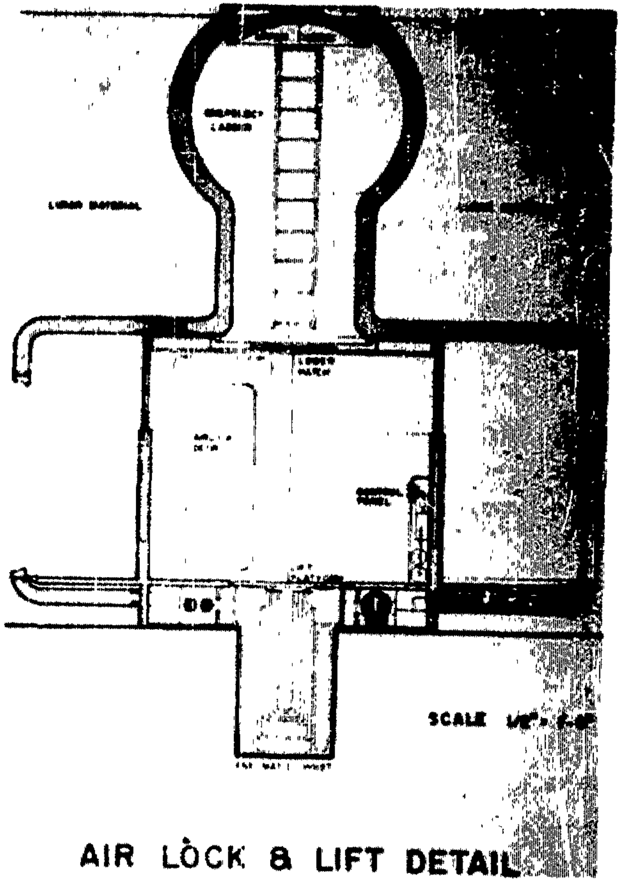


FIGURE 14

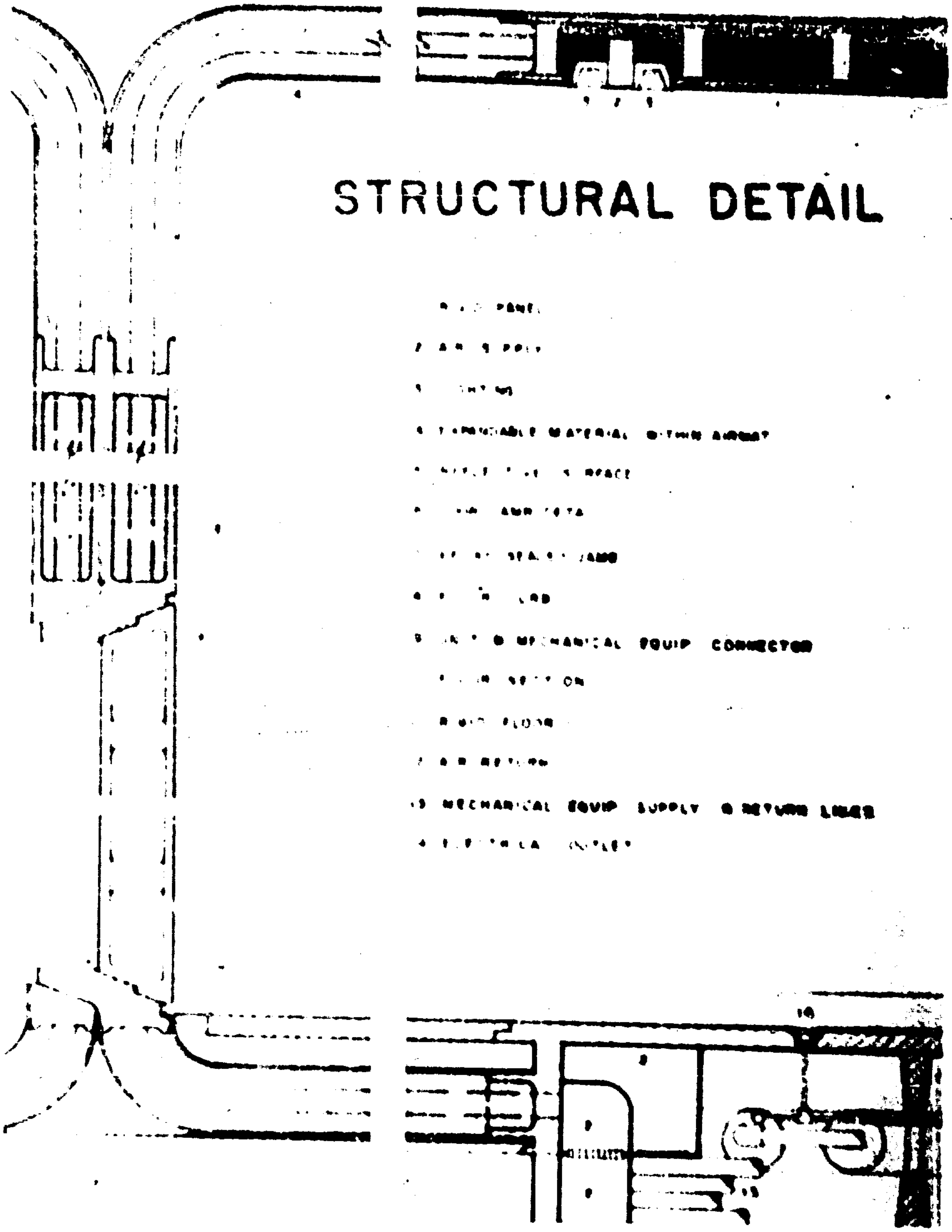

STRUCTURAL DETAIL
MECHANICAL EQUIP CONNECTOR
MECHANICAL EQUIP SUPPLY

FIGURE 15

Designers:
T. Berkhouse, Ind. Des.
F. Conboy, Ind. Des. '[illegible]
P. Franz, Arch.'64
G. Hoffmann, Arch. '64

PA[illegible]

The package technique [illegible] in the hull of the space sh[illegible] equipment necessary to sust[illegible] packaging feature is that t[illegible] of 38 inches each and still [illegible] thus eliminating the job of [illegible] hut after expansion.

The first step toward [illegible] nose from the space ship an[illegible] the lunar roving vehicle. [illegible] huts are "plugged" into the [illegible] solar collector and drop th[illegible] ship hull, thus forming th[illegible]

The huts are expanded [illegible]

The space left betwee[illegible] with balloon-shaped foam.

After everything is in[illegible] lunar "stockade" is formed [illegible] radiation and micrometeorite [illegible] and the ceiling line determi[illegible] the stockade.

STRU[illegible]TURES AND MATERIALS

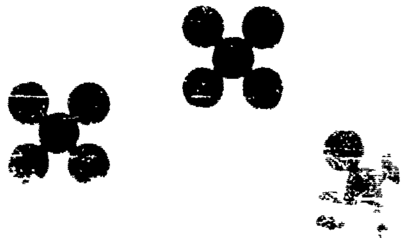
The structure in the in[illegible] expandable huts is first provided by interior air pressure and then by [illegible]ated hardened foam.

The 34 inch bases of the huts are of a double wall construction of the same type as used in the space ship hull, while the upper inflatable membrane is covered with micro-incapsulated foam reactants. After the huts are positioned in place and inflated, the capsules are melted away and the foaming action is initiated.

The structure in the ceiling covering of the stockade is made up of ribs of the space ship hull. At landing time these ribs form the upper 23 feet of the shell of the space ship hull. After the huts are in place, these ribs fall away on hinges at the lower ends. These ribs are held in a near-horizontal position by tension rings, and the voids left between the ribs are filled with balloon-shaped foam. The lower 12 foot portion serves as the central structural column for the umbrella-type stockade ceiling.

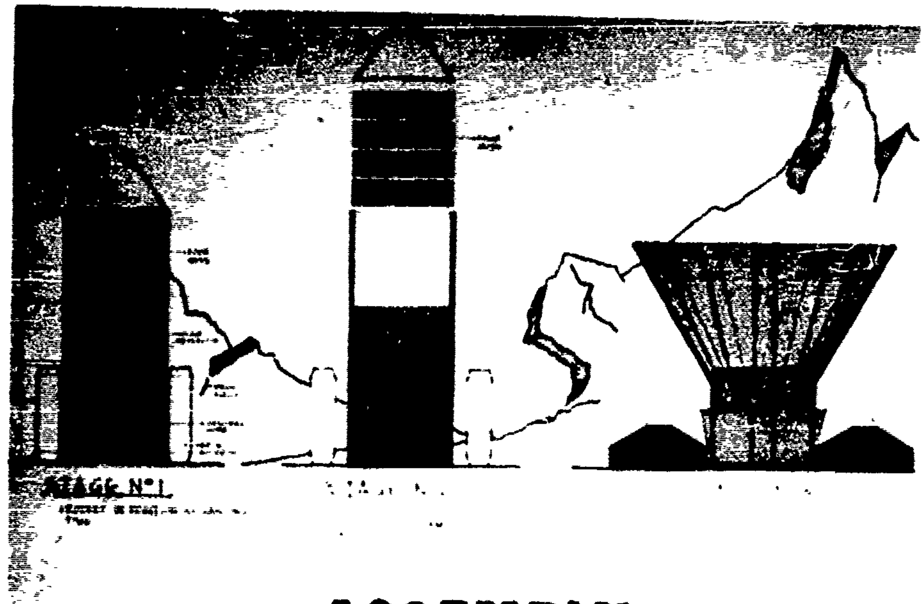
Lunar dust is used as a protective covering on the roof of the stockade and by forming walls around the lunar shelter.

LUNAR SHELTER



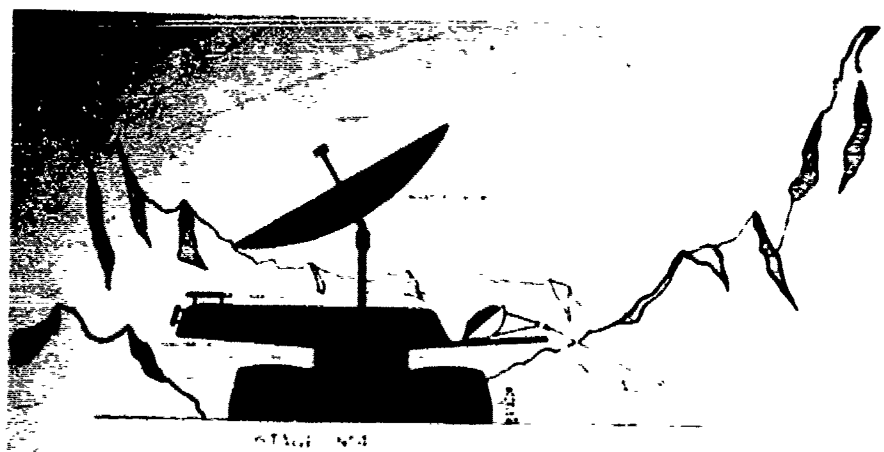
COLONY

PLAN



ASSEMBLY

FIGURE 18



ASSEMBLY

FIGURE 19

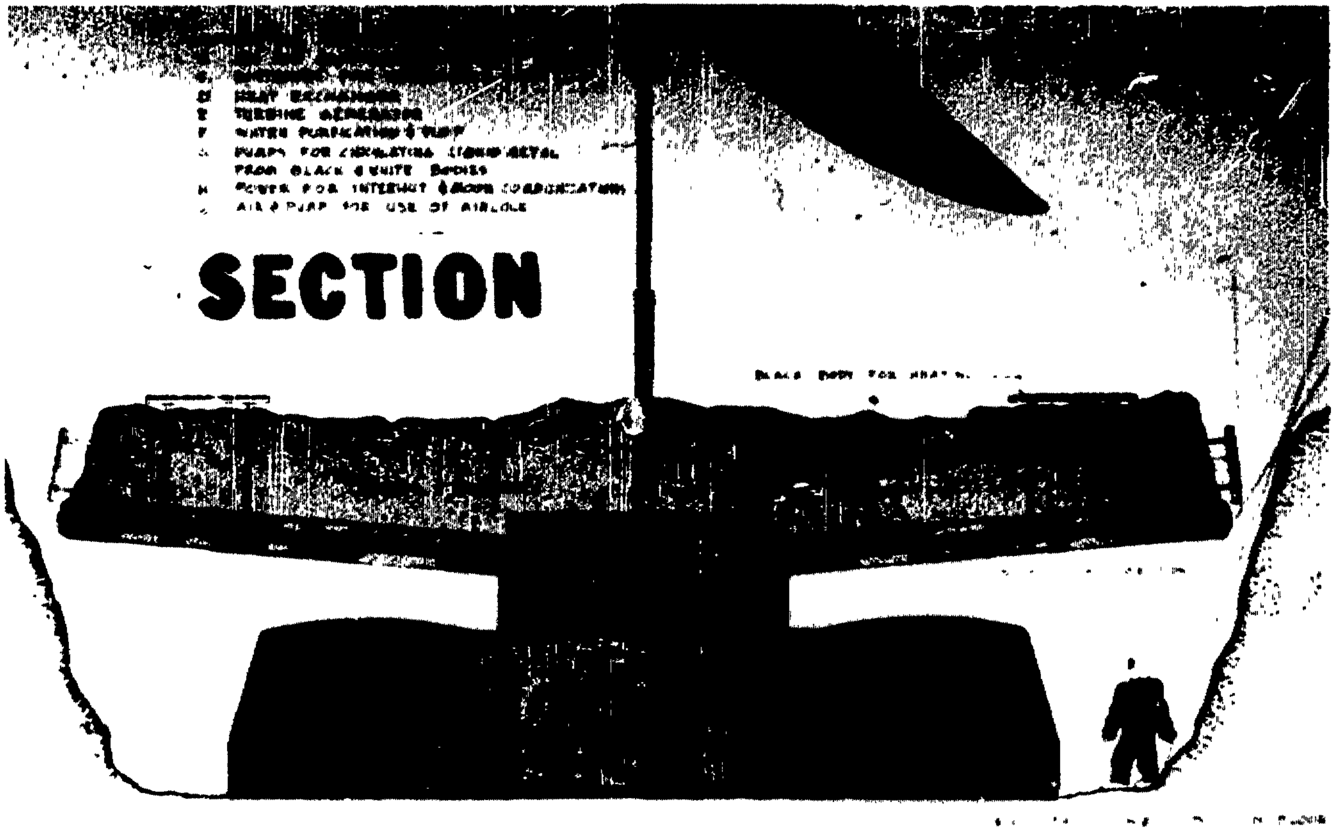


FIGURE 20

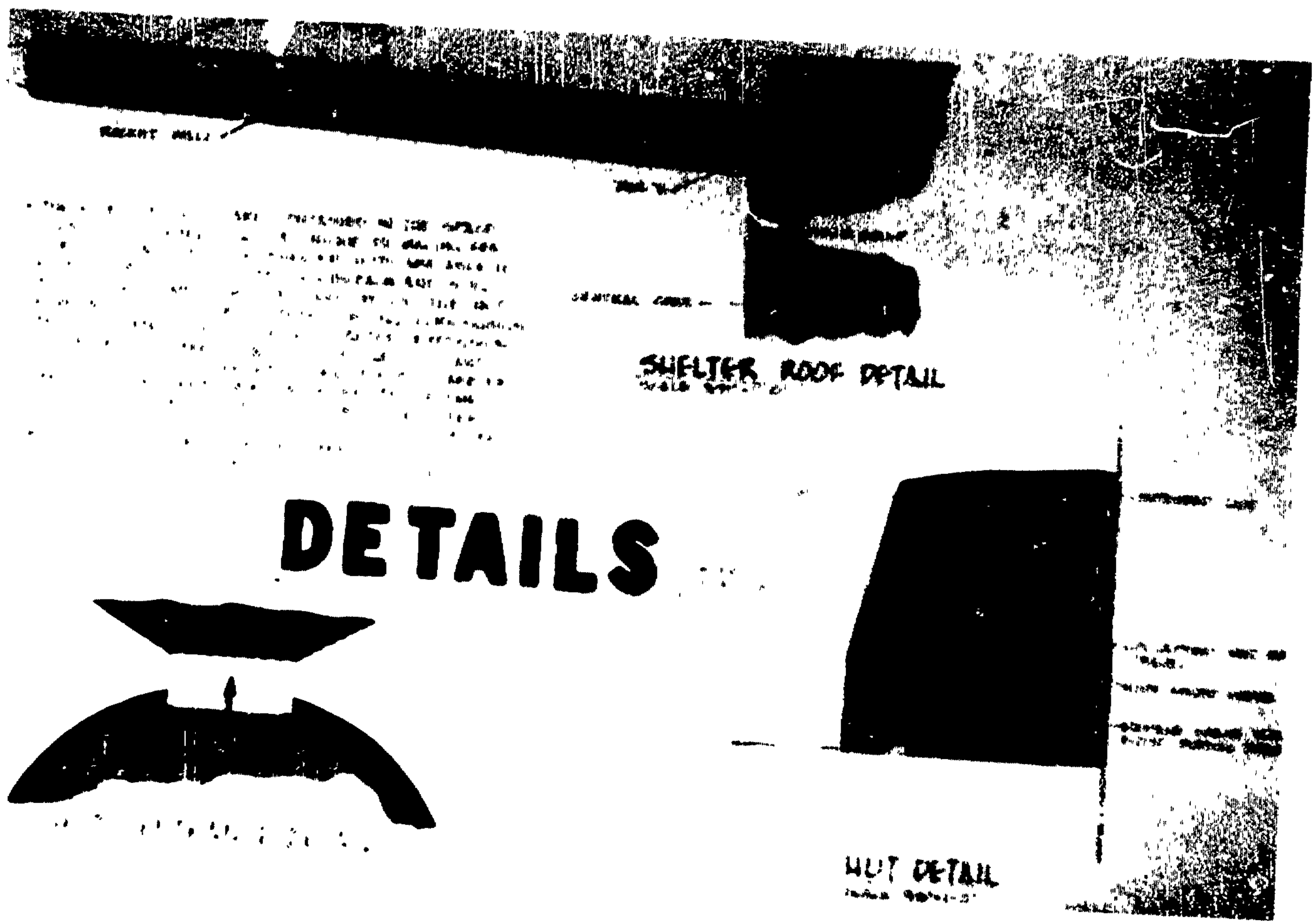
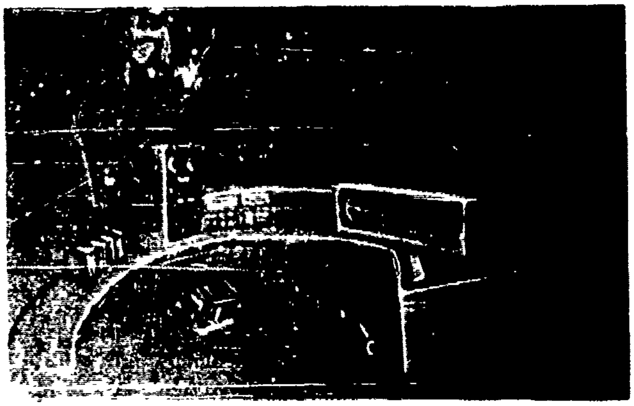
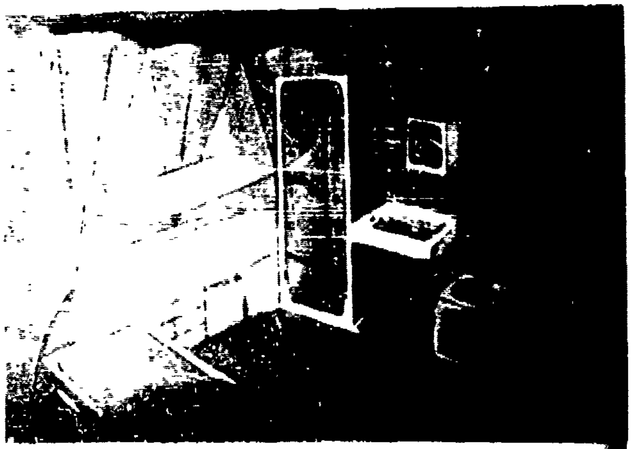


FIGURE 21



LAB COMMUNICATION



SLEEPING

FIGURE 22

FIGURE 23

FIGURE 24

SOLUTION OF TEAM #12

Designers:
H. DeBaun, Ind. Des.'63
J. Wells, Arch. '64
J. Wyler, Arch '64
R. Yelton, Arch. '64

PACKAGE AND ERECTION TECHNIQUE

The rocket will be sent directly from earth with a lunar package thirty-five feet in length by fourteen feet six inches in diameter. The upper five feet will house pod rockets with adjustable legs capable of securing the rockets in the vertical position. This upper portion contains the solar reflector, heat transfer equipment, communications equipment, and television cameras.

This tower-like structure will also act as a gantry for lowering the thirty-foot shelter (with the aid of the lunar vehicle) into the appropriate horizontal position in a hole produced by explosives fired from the descending rocket. At placement, the package would automatically be split open, separating into equal halves ready for habitation. Leg-like structures will then level the shelter in preparation for covering with lunar substance. All cables and lires would be already connected to the gantry. This whole procedure would take a minimum amount of time saving the men for more useful research.

STRUCTURES AND MATERIALS

The lunar shelter is to be completely prepackaged, supplied, and built on earth using the capsule shell of the package as the structural members. This shell will be filament-wound asbestos-phenolics. The cylindrical shell will fuse into equal halves using silcone - hydraulics to separate the sections nine feet apart, with solid insulated folding partitions expanding simultaneously with the division of the shells guaranteeing always an earth-type environment within the station.

FIGURE 25

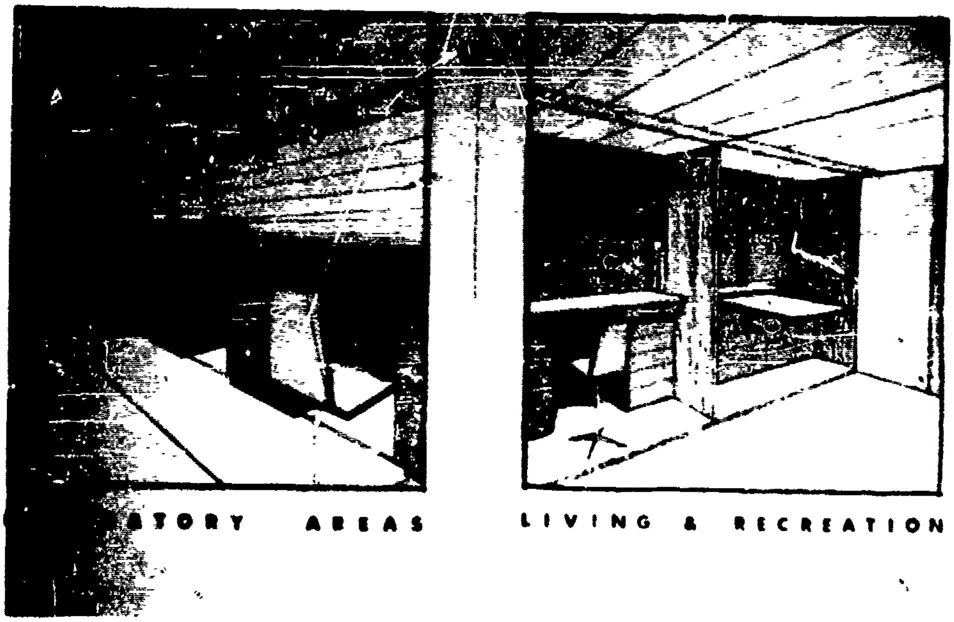
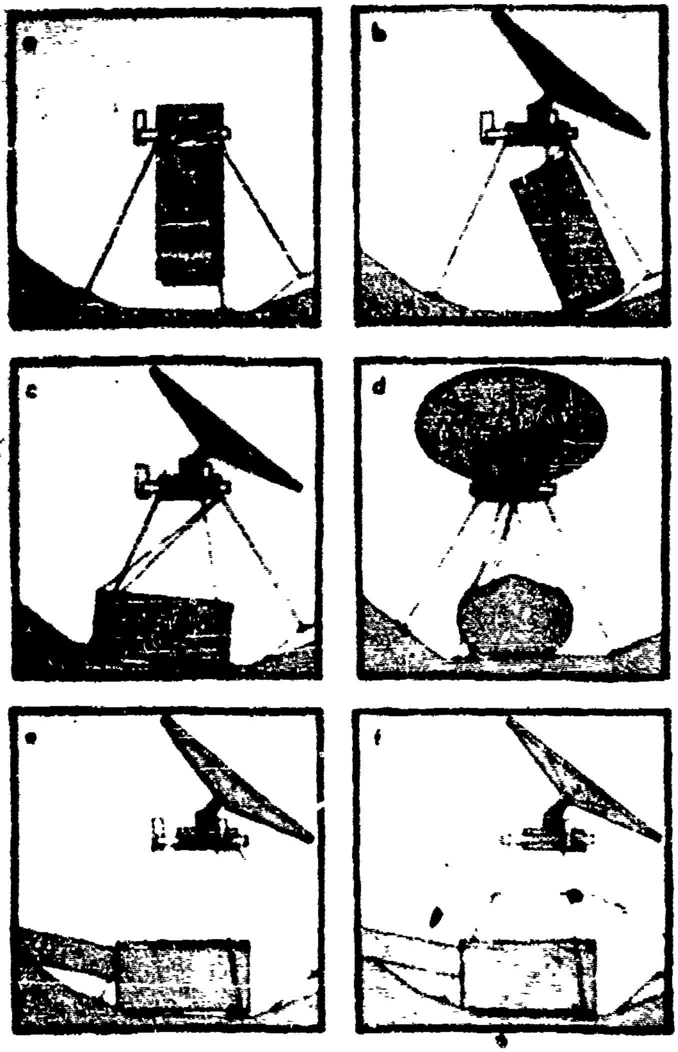
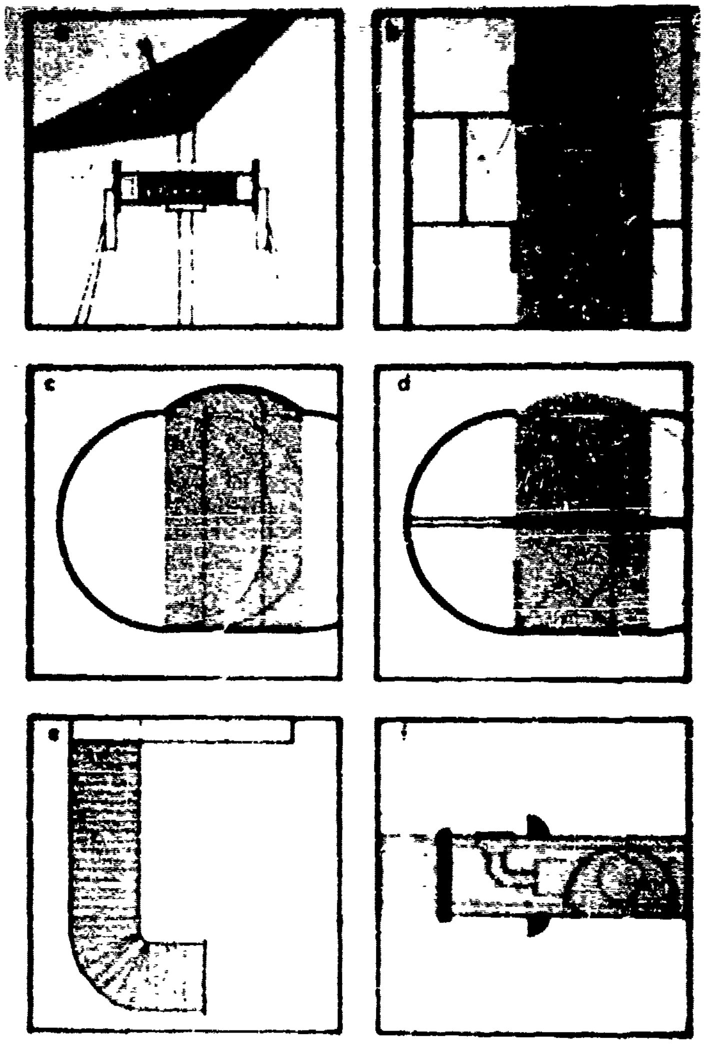


FIGURE 26



ERECTION DETAILS

FIGURE 27



E R E C T I O N   D E T A I L S

FIGURE 28

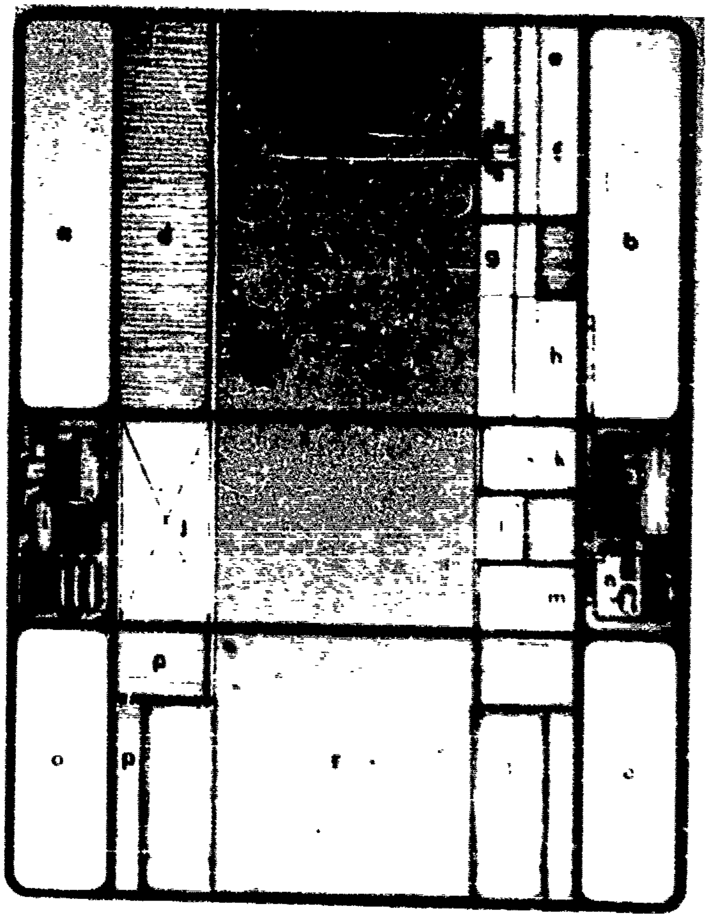


FIGURE 29

LOWER LEVEL PLAN

- a. air storage
- b. water storage
- c. living area
- d. recreation storage
- e. first aid equip.
- f. study area
- g. food preparation
- h. food storage
- i. air purification redistribution equipment
- j. elevator
- k. shower
- l. lavatory
- m. water closet
- n. waste purification
- o. air storage
- p. personal storage
- q. bunks
- r. sleeping area

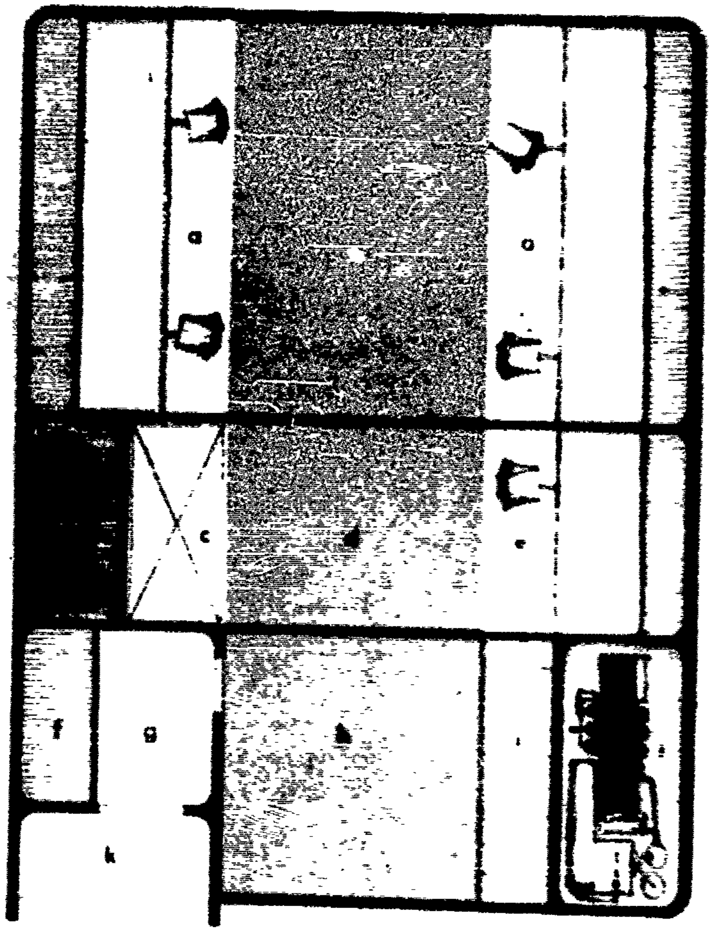


FIGURE 30

UPPER LEVEL PLAN

a. laboratory areas
b. emergency power
c. elevator
d. circulation
e. communications control
f. exterior equipment
g. airlock
h. changing area
i. space suit storage
j. thermal control
k. tube to surface expanded

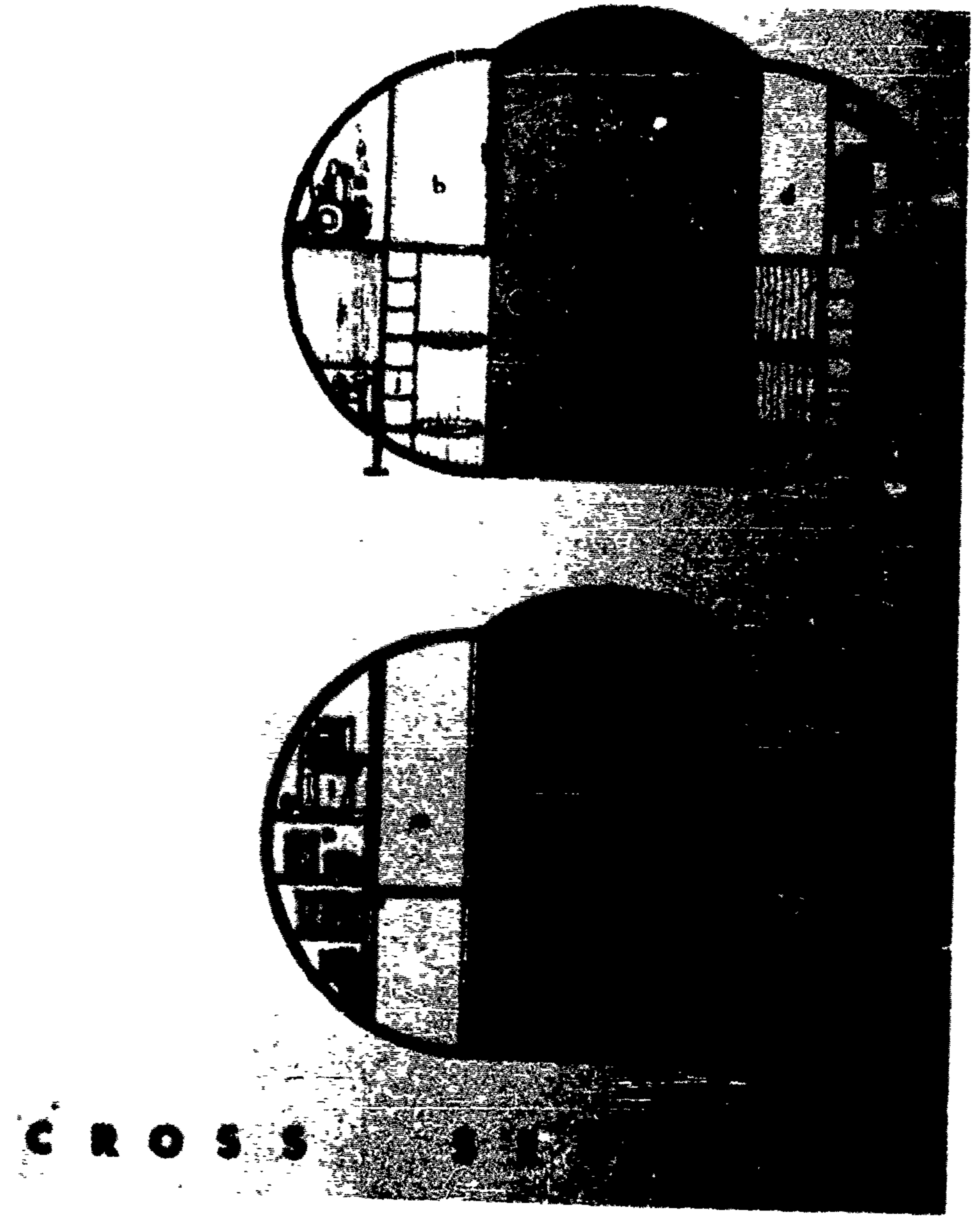


FIGURE 31

NASA RESEARCH ON EXPANDABLE STRUCTURES

Norman J. Mayer

Office of Advanced Research and Technology
NASA Headquarters, Washington, D.C.

SUMMARY

Expandable structures have been or may be utilized in space missions as passive communication satellites, antennas, solar power systems, recovery and reentry systems, space laboratories, lunar bases, and many items of equipment. Research in materials and structures is being directed towards solving the common problems in these applications and to advancing the technology. Special emphasis should be given to non-metallic materials in filamentary form. More research is needed to broaden the field and develop the potential that expandable structures possess.

INTRODUCTION

The choice of methods for accomplishing the United States mission of the peaceful exploration of space includes many programs and projects. Most of the programs are probing the unknown frontiers in science and they often strain the state-of-the-art in vehicle technology for their accomplishments. The provision of feasibility sometimes requires a drastic departure from traditional structural approaches.

Once feasibility is attained, the resulting system or concept must be evaluated in terms of the reliability and efficiency needed for the mission. It is because structures of variable geometry or expandable form have provided solutions which satisfy these three requirements that the NASA program includes both a number of applications for expandables and continuing related materials and structures research programs. In some cases expandable structures have provided the only practical solution. In order to provide technology in a timely manner for future requirements for expandable structures, research programs have been established often on the basis of choosing hypothetical future missions. These establish a frame of reference for applied research even though the mission itself may never materialize. These projects also serve to point out items such as material developments and analytical methods which will be required in general for future expandable space vehicles.

It has been found that expandable structures are chosen under one of these three conditions:

1. When loads and stresses on the launch vehicle or payload structure can be reduced significantly during launch by stowing or packaging.

2. When the mission dimensions of the payload structure are large enough to critically affect the aerodynamics, dynamics, or stability of the launch vehicle.

3. When the structure must be housed or stowed but sufficient volume is not available in the system for such containment.

Most spacecraft possess some portion or component which is deployed in space. For the purposes of this report, however, the category of expandables is limited to those cases where a major portion of the structure is deployed or expanded. This includes systems which unfold, extend, unroll, or inflate. Such characteristics dictate use of materials and methods which allow changes in geometry without loss of physical properties necessary to the remainder of the mission.

Structures with these characteristics fall into two groups: those permitting expansion through rigid elements which telescope, hinge, or unroll, and those which utilize the characteristics of very ductile materials for expansion. It is the purpose of this paper to describe NASA research in expandable structures and the interrelationship between research and applications.

APPLIED RESEARCH ON EXPANDABLE STRUCTURES

Expandable structures have been or may be employed for the following applications: passive communications satellites, antennas, space power systems, reentry and recovery systems, manned orbital laboratories, lunar bases, and various items of equipment.

Passive Communications Satellites - Expandable structures have served well in application to passive communication satellites. The Echo I, still in orbit, is an excellent example of the case where the feasibility of orbiting a 100-foot diameter sphere would have been unattainable except by use of a thin-shelled expandable design.

Research continues in passive communication systems because of their relative invulnerability to either the space environment or man-made interference. They will always require large dimensions and therefore may be expandable.

There are several critical problems encountered in building

and operating these structures, since they encompass a different class of material form and geometry than that involved in smaller compact spacecraft. Some of these problems have been reviewed in Reference 1. In general they are primarily problems of rigidization and deployment.

The unique operational requirements produce structural problems as well as operational problems. In achieving proper shape, methods must be chosen to stiffen the shell. The combination of pressurized deployment and thin films alone has not been entirely satisfactory. The second Echo, the A-12 system shown in Figure 1, incorporates the feature of a laminated aluminum-Mylar skin yielded by internal pressure to final spherical shape. Studies (Reference 2) have indicated that a system of less weight than the A-12 system is attainable. Aluminum foil in thicknesses approaching 0.1 mil appears to be practical. Polypropylene, a lighter film, may be substituted for Mylar. Other new polymer developments are being considered which could lend themselves to improved satellite structures. Lamination of metals and plastics could be avoided by using an all-plastic film system. A reliable method of rigidizing the film after deployment must be developed along with the required RF reflective surface. Recent research on UV and IR activated rigidizing systems points the way towards achieving this objective. Further developments, as described in Reference 3, in radiation activated cross-linking in polymers promise another possibility for non-metallic systems.

Research on advanced passive systems have shown that the numbers of satellites required for essentially continuous world-wide communications may be reduced by an order of magnitude if precise orbiting positions can be maintained. This can be achieved by eliminating the effects of external perturbing forces. The perturbing forces are primarily solar pressure effects. Suggested solutions represent some interesting structural developments. One approach under investigation by NASA employs an open metal grid for the main structure. A scale model is shown in Figure 2. In this case, the entire reflective structure is formed of 2 mil aluminum wire. The grid structure is covered with a polymeric film. This forms a sealed container which can be expanded by the internal pressure produced by a subliming substance or other means. After expansion, the film coating is automatically removed by the process of photolyzation, leaving the open grid structure. Such structure presents a much reduced exposed area for solar pressure action.

The lenticular shape shown in Figure 2 was selected for study on the basis of a 4-5 db gain in usable signal strength over that attainable by completely spherical arrangements of similar dimensions. The structural arrangement of such shapes requires that a peripheral ring or torus be added to provide the desired reflector contour. The tripod structure is also expandable and supports a gravity-gradient stabilization system.

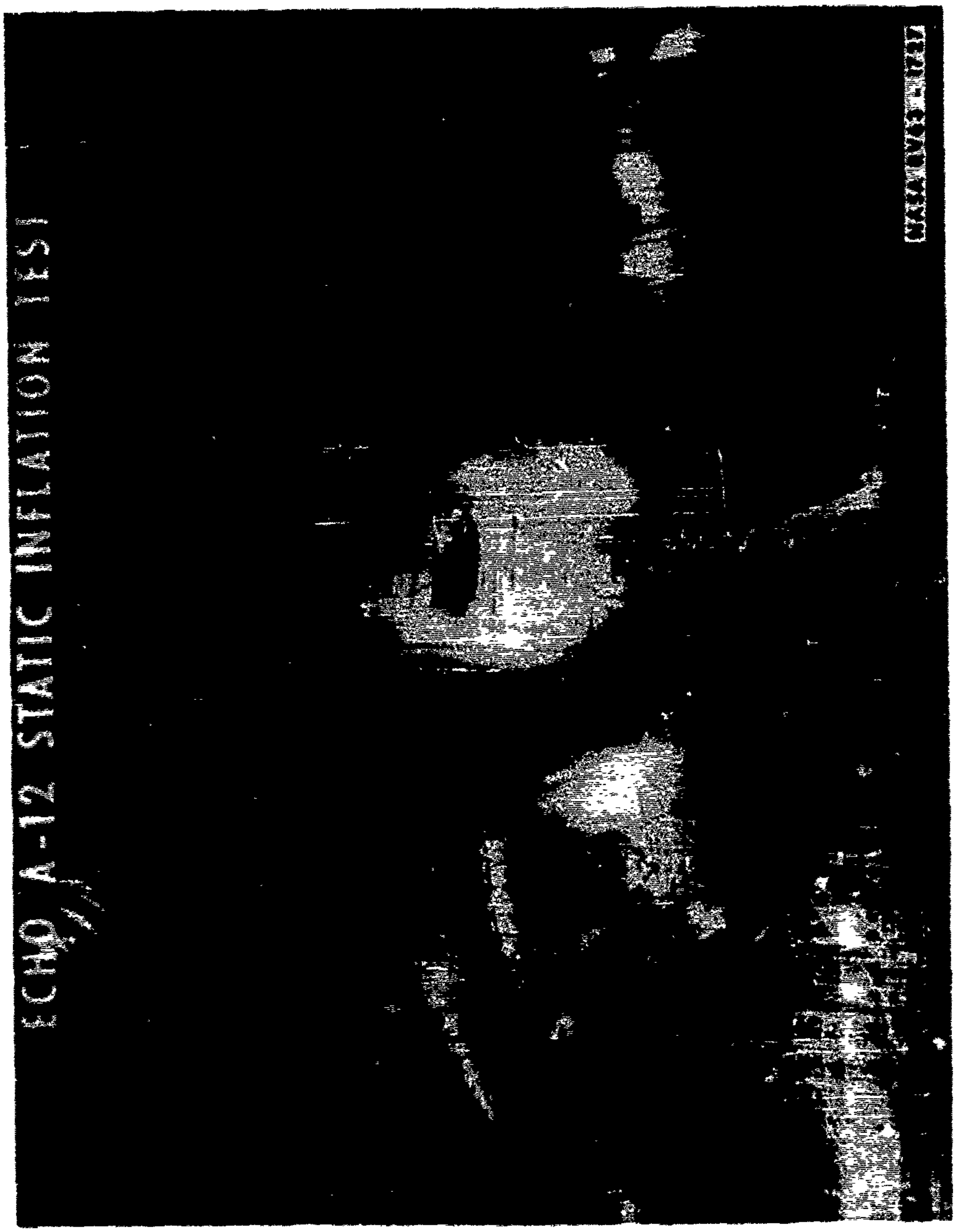


Figure 1 - Echo A-12 Static Inflation Test

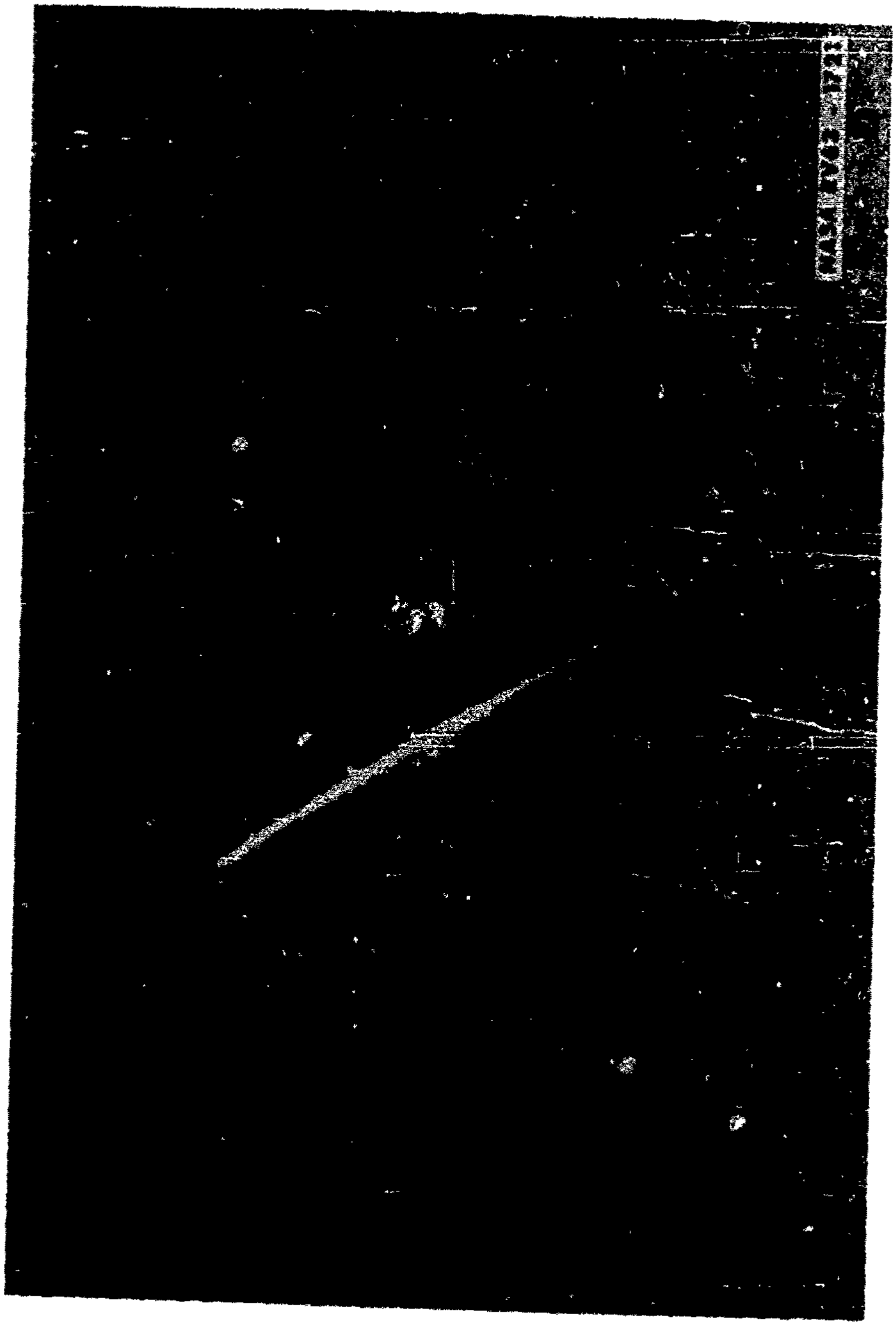


Figure 2 - Lenticular Communications Satellite

Another approach being studied retains the sphere but utilizes solar pressure effects by orienting properly coated panels towards the sun. The orientation is provided by magnetic meridional coils on the satellite. The change in orientation produces orbital changes through solar pressure action to compensate for original perturbations. All of these studies will provide a technological basis for choosing an optimum system should one be required in the future.

Antennas - Radio and radar antennas for use in space benefit from being packaged for launching and later expanded in space. By eliminating aerodynamic loads, many light-weight structural concepts become feasible. One example is shown in Figure 3. This is the Yagi-Disc arrangement which has been investigated by NASA. Figure 3 shows the concept in its packaged and expanded condition. The discs are positioned between blocks of polyurethane foam which are compressed for launching. In space, the package is released and the elastic foam provides the erection and spacing for the antenna discs. A description of the experimental research on this system is reported in Reference 4.

Very large expandable antennas useful for tracking and radio-astronomy work in space are also being studied on the basis that future missions may justify systems of such magnitude. Figure 4 shows one system under study. Over-all dimensions for some variations of this concept may reach 10,000 feet, and the need for novel structural approaches such as expandables is evident. It is anticipated that such research will benefit other large structures as well.

Space Power Systems - Many of the various sources of electrical power in space require the exposure of large surfaces to the sun. Some power requirements are small enough to allow utilization of the surface of the payload package or space vehicle for mounting of photovoltaic silicon solar cells, but future systems producing as much as 20-30 kw of electric power may require very large surfaces (Reference 5). If the solar cell system is employed, flat panels are used to mount the cells. These are folded for launching and deployed in space. The reliability and freedom from critical geometry requirements continue to make this arrangement attractive for many systems. Another solar cell concept is a so-called "rug" configuration, which eliminates flat metal panels. In this case cells would be mounted on an elastomer coated cloth structure which is rolled up for packaging. It would be stiffened after deployment.

Thermionic systems, such as parabolic concentrators, represent considerable weight saving over the solar cell concept when large amounts of power are required. These are discussed in Reference 6. Figure 5 shows values for power produced by two systems, which represent possible upper and lower bounds of efficiencies, plotted

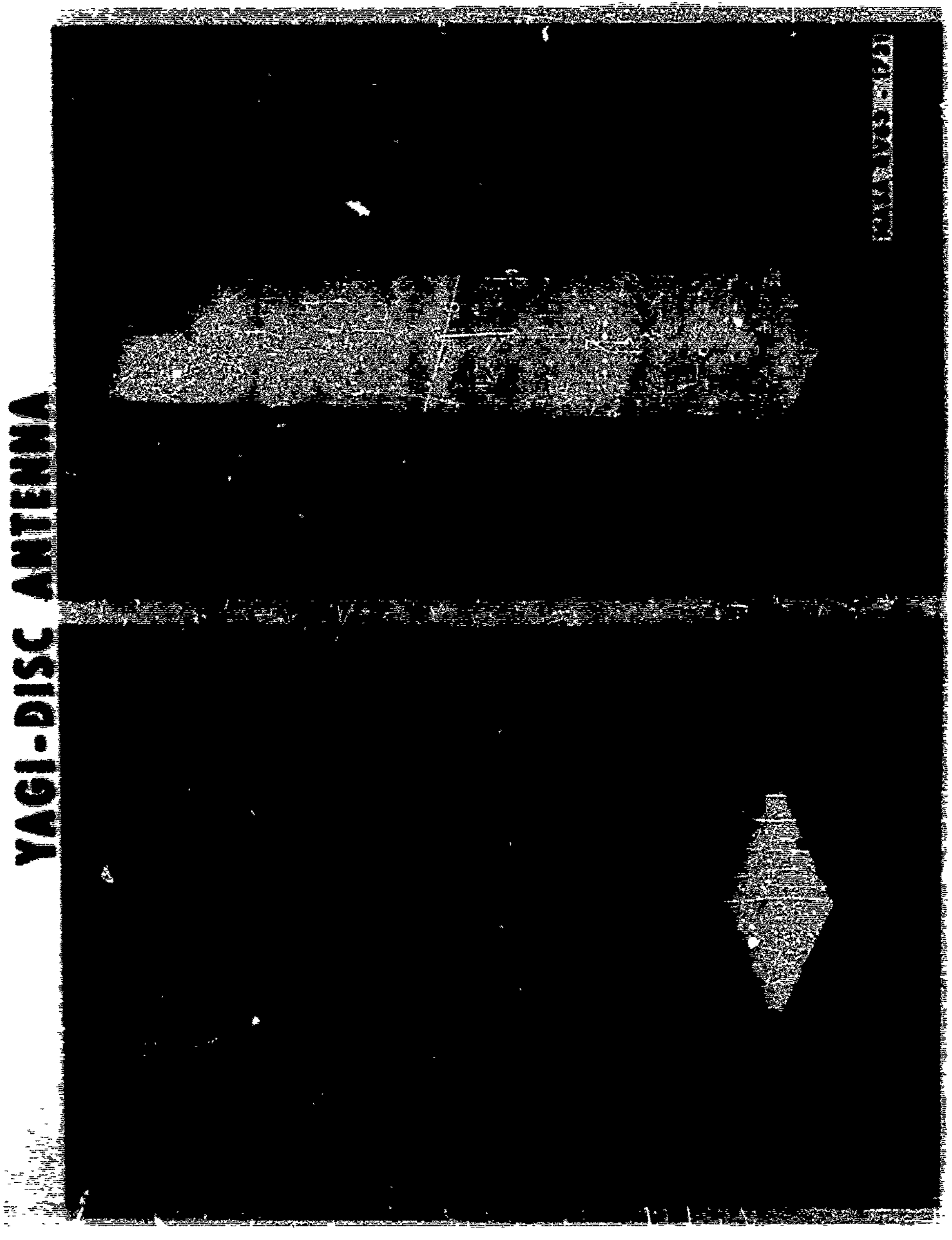


Figure 3 - Yagi Disc Antenna

Figure 4 - Orbital Radio Telescope

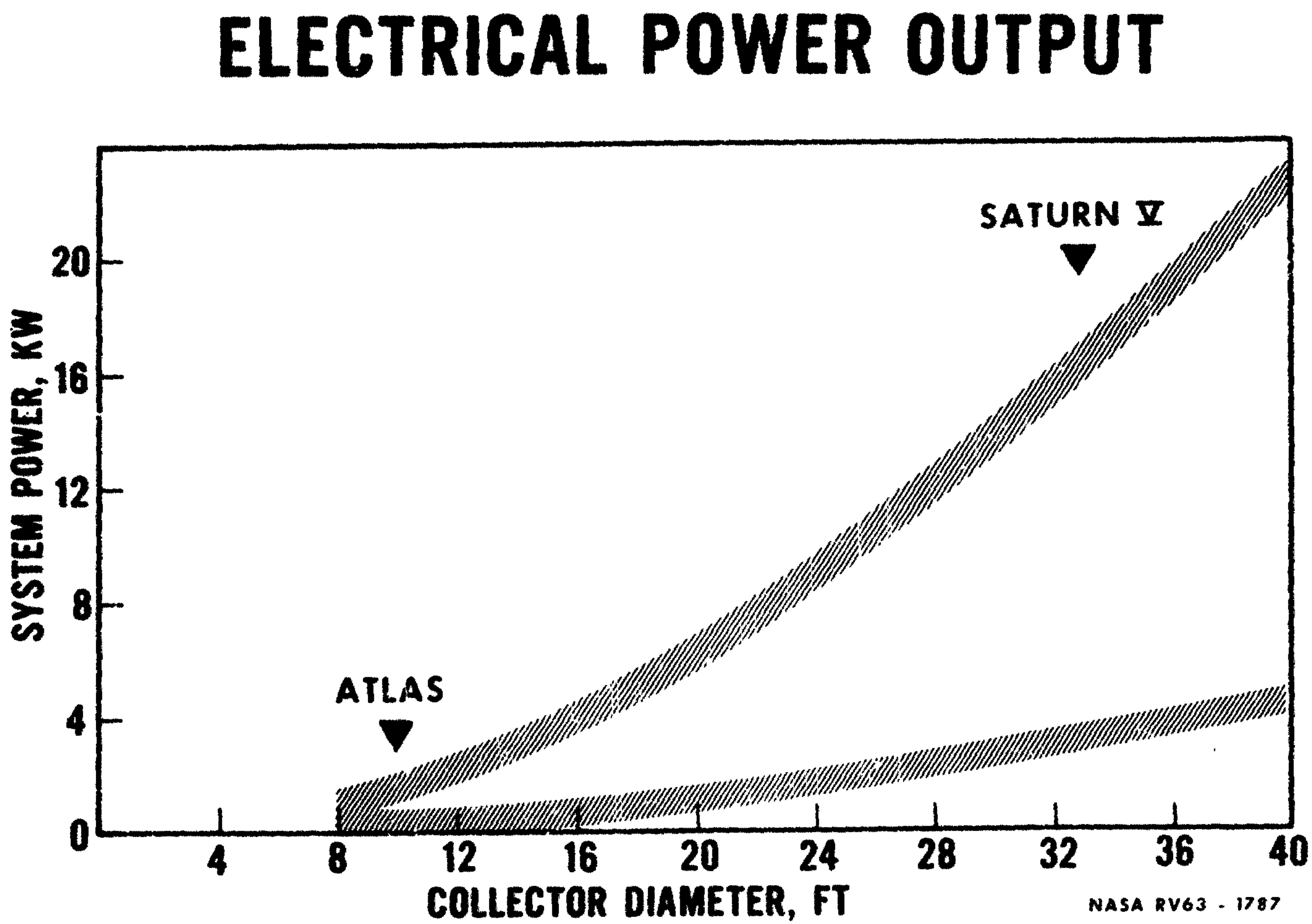


Figure 5 - Collector Diameter Versus Power Output

against collector diameter. It is obvious that diameters quickly exceed present launch vehicle package dimensions and the need for expandables is quite clear. Among the thermionic systems, there are also choices which produce minimum weight. Figure 6 shows a plot of specific power versus the operating temperature of the concentrator absorber for several systems. All of the systems shown are expandable types.. Data shown are based on laboratory models. It is possible and probable that the relative positions of some of the systems shown may change with actual space operating requirements. For example, the inflated system would require continuous pressurization. The total weight of gas required would vary with the total mission time. Figure 7 shows that the pliant types permit smaller packages for launch. Three recent examples which have been studied utilizing pliant concepts are:

1. A parabolic reflector constructed of aluminized film. It is deployed by a system which inflates both a peripheral ring and a temporary hemispherical cover. A rigidizing process would be used after initial deployment to render the system free of the continued need of pressure. A model of this reflector is shown in Figure 8.

2. A spin deployed reflector. This is an arrangement where centrifugal force is used to deploy the reflector. The reflector is a performed Mylar paraboloid. A cable system is used to maintain the desired peripheral reactions. Figure 9 shows a model under test in a vacuum chamber.

3. So-called umbrella types have been studied and some of this work is reported in Reference 7. Figure 10 shows a 10-foot diameter model of one of these systems.

Common problems exist among all systems. These include:

1. Achieving and maintaining accurate dimensions.

2. Maintaining reflector surface brightness.

3. Protection against material deterioration and damage due to space environment.

Other more compact systems, such as nuclear power, do not completely eliminate the necessity of requiring large surfaces as a part of the system. The great amount of waste heat which must be displaced leads to large areas for radiators. At the present time, no concept exists to completely satisfy the severe structural problem of providing radiation surfaces in space. Present approaches involve the prosaic solution of fluid flow through pipes. Such systems would employ a considerable number of furlable components to deploy the radiators after launch. This area is one where

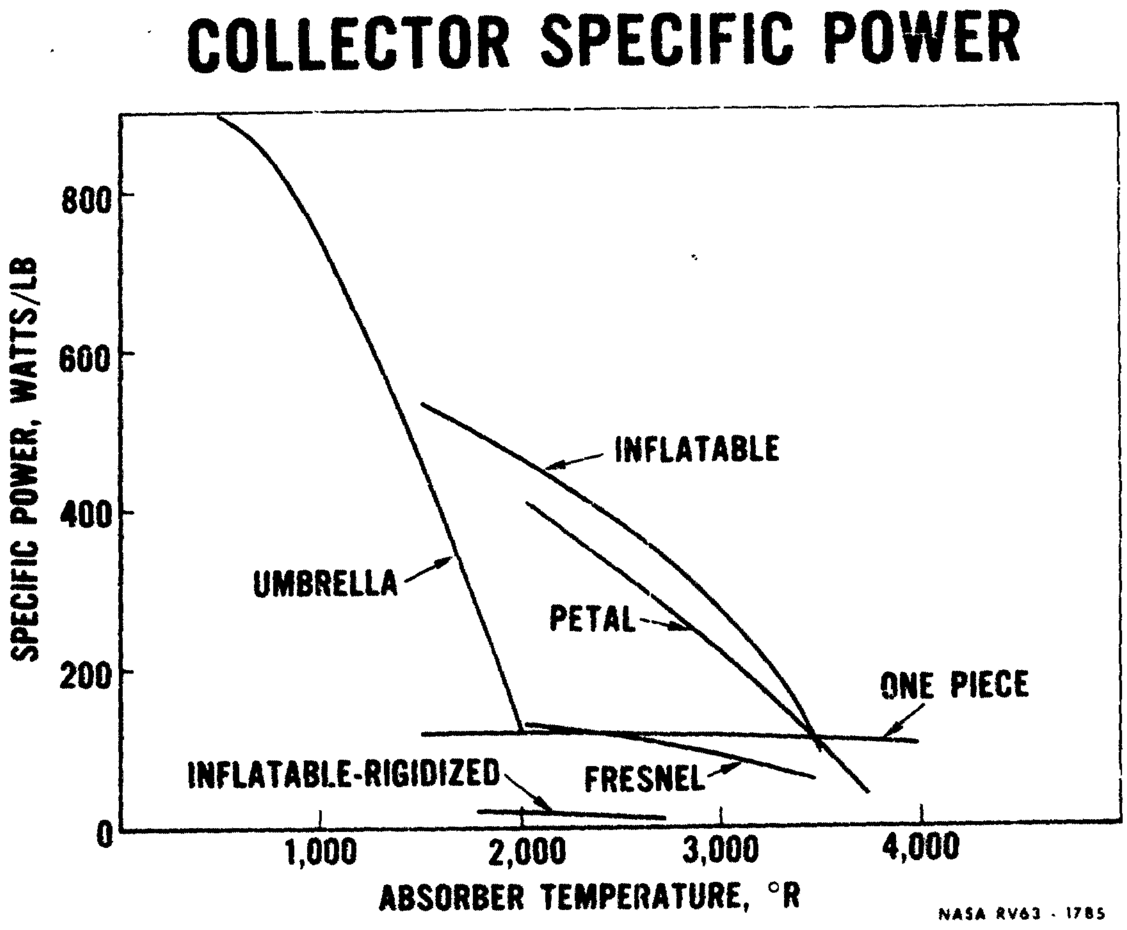


Figure 6 - Collector Specific Power Versus Temperature

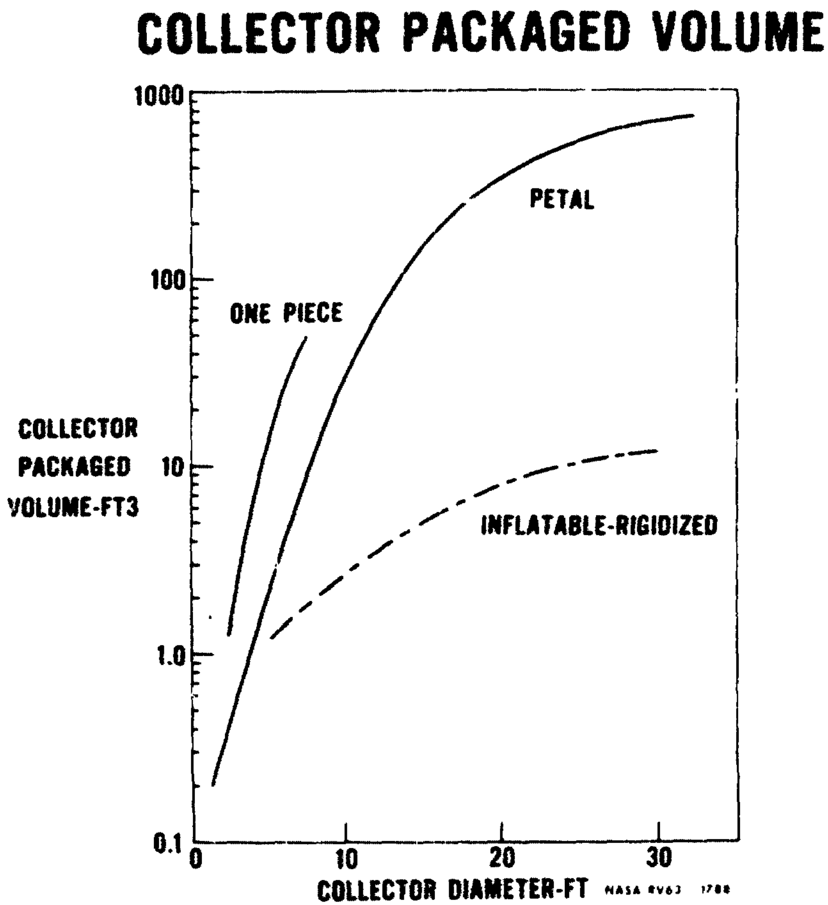


Figure 7 - Collector Diameter Versus Package Volume

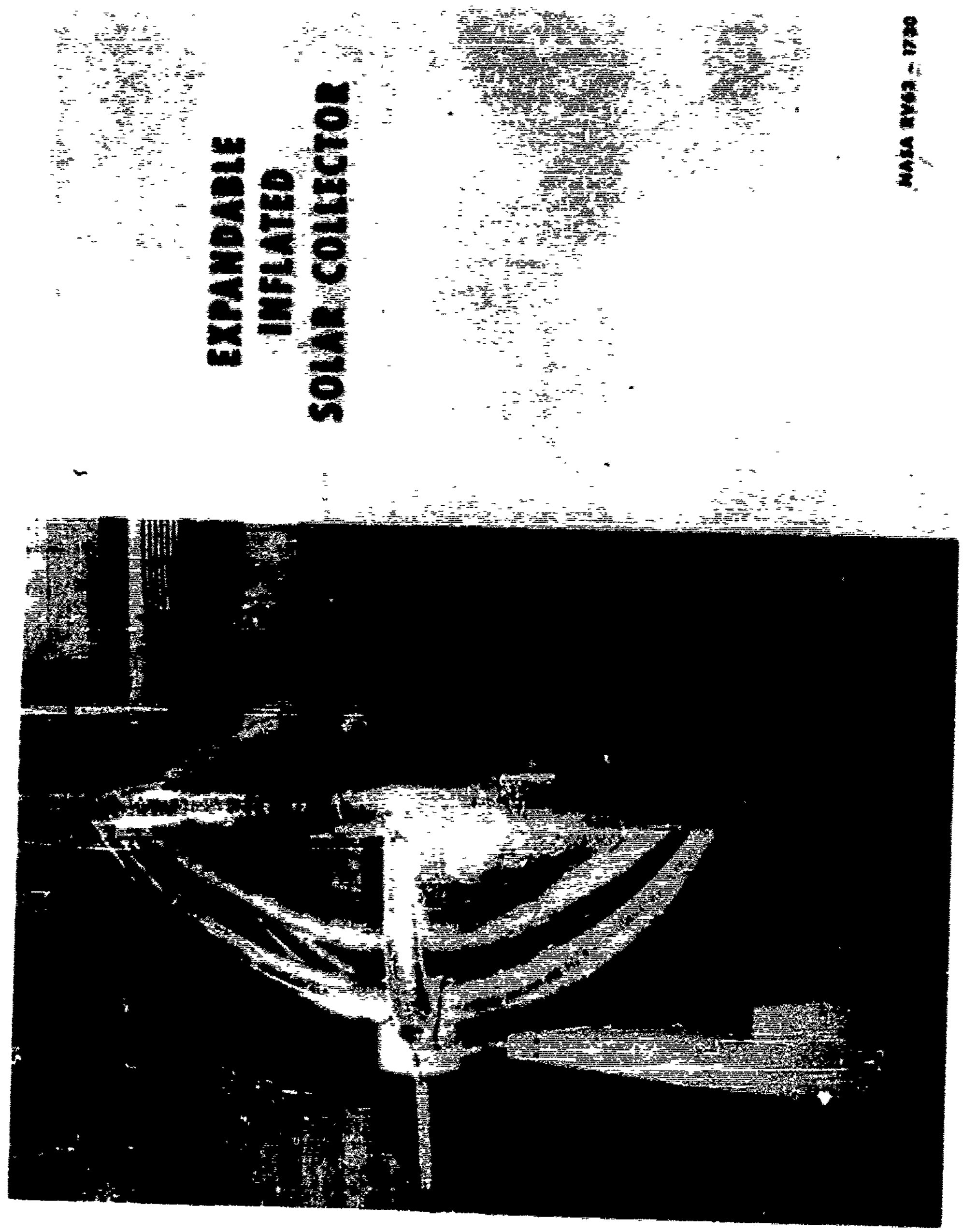


Figure 8 - Model of Inflated Solar Collector

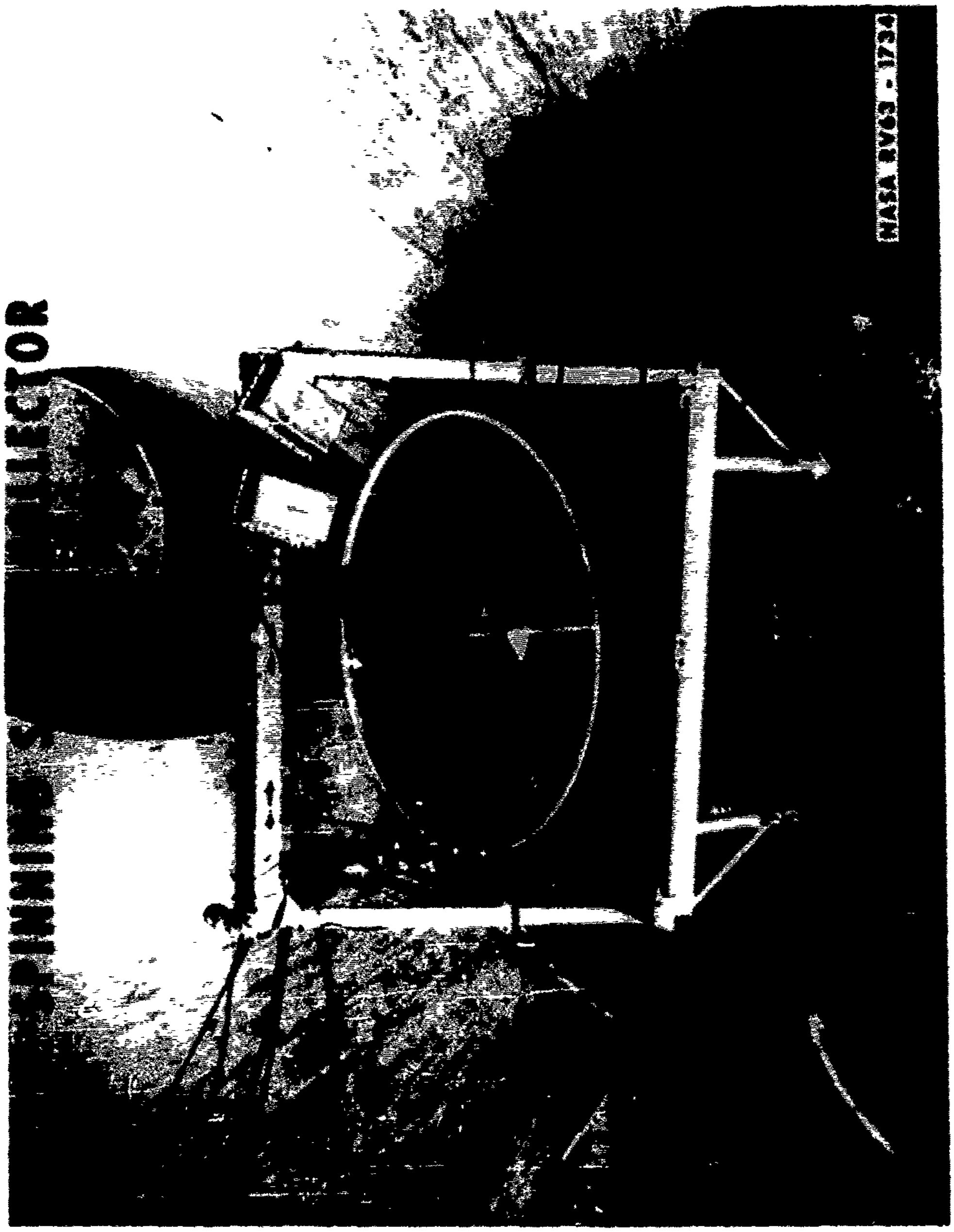


Figure 9 - Spinning Solar Collector

Figure 10 - Umbrella Collector

inventive ability and imaginative approach is needed. A departure from traditional heat exchange solutions to circumvent the problem may be necessary.

Reentry and Recovery Systems - The reentry and recovery of spacecraft is one of the most important phases of spacecraft operation. The recovery phase has been handled by various parachute systems up to the present time. Advanced systems will attempt to provide a glide and control capability to allow selection of a landing site, such as the use of the deployable paraglider on Project Gemini. More advanced systems are being studied in attempts to combine reentry and recovery. An example is the M series of glide reentry vehicles. In this study, the NASA has looked at three variations of a similar configuration. One of these, the M1-L body, is a semi-cone shape which reenters using ablation for thermal protection. After reentry, an afterbody portion would be deployed. This portion would contain stabilizing and control surfaces and be constructed of both plied coated fabric and panels of dual-walled fabric. The inflated portion of the system was chosen on the basis of its weight being equivalent to that of a parachute. Maximum temperatures of 300°F are expected during the initial recovery phases. A large-scale test model of this concept is shown in Figure 11.

Another M type body is the M-3 which features wings which are folded during initial reentry. These are later unfolded for gliding and landing. Later versions could utilize powered flight as well. This system is discussed in Reference 8.

The NASA is also investigating combined reentry and recovery from the standpoint of using centrifugally deployed variable geometry surfaces. By controlling the apex angle of a deployed cone or vane system, the concept of "iso-thermal" flight or atmospheric entry at constant surface temperature is utilized. Efficiency studies of this system indicate promise of substantial weight saving over other systems including ablative reentry. Figure 12 shows one form the system could take. As shown, the rotors are monotropic filamentary structures and susceptible to very compact packaging. Figure 13 shows a comparison of materials that could be utilized as filaments in the blade structure. Q is a materials property factor involving emissivity, temperature and strength. Reference 9 contains a discussion and analysis of the iso-thermal concept. This study is continuing with special emphasis on the use of boron filaments. Other filaments will also be investigated.

The expandable glider principle is being employed in another form to obtain scientific information. This is a micrometeoroid project which involves the launching of a paraglider in a near vertical trajectory at a maximum altitude of 700,000 feet. At this altitude the glider is deployed by automatic inflation of its sail

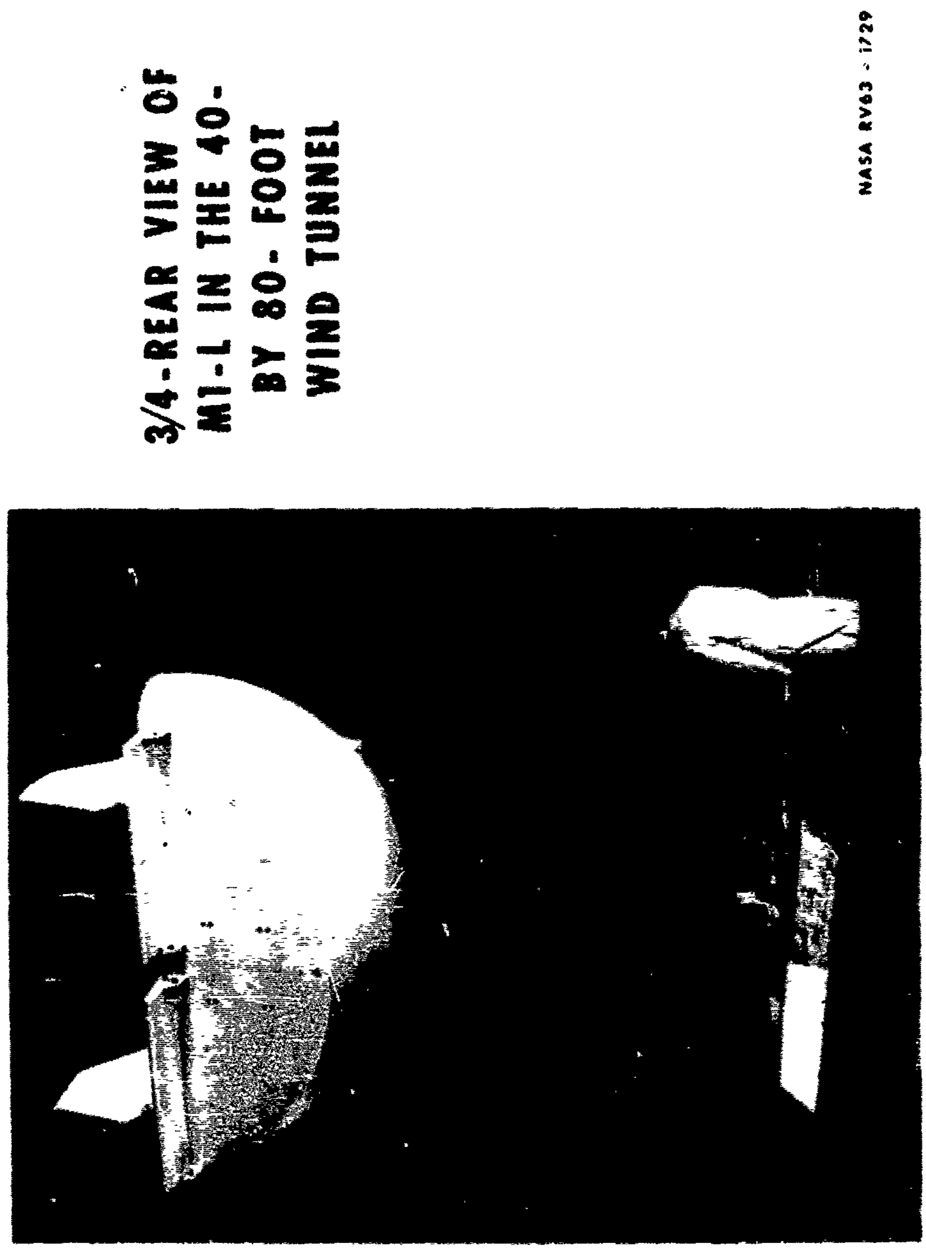


Figure 11 - M1-L Reentry Vehicle

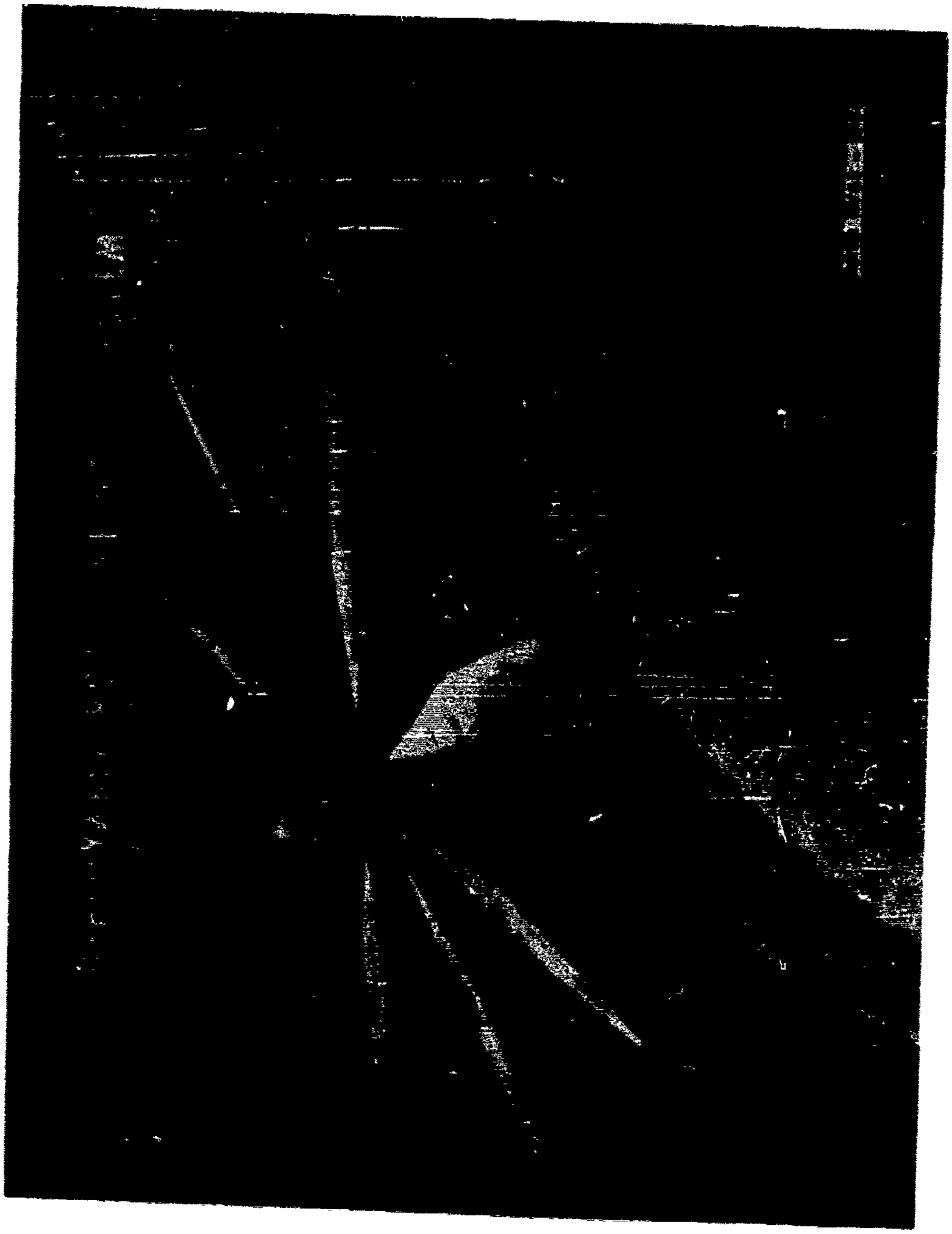


Figure 12 - Deployable Reentry System

MATERIALS PARAMETER Q VS TEMPERATURE

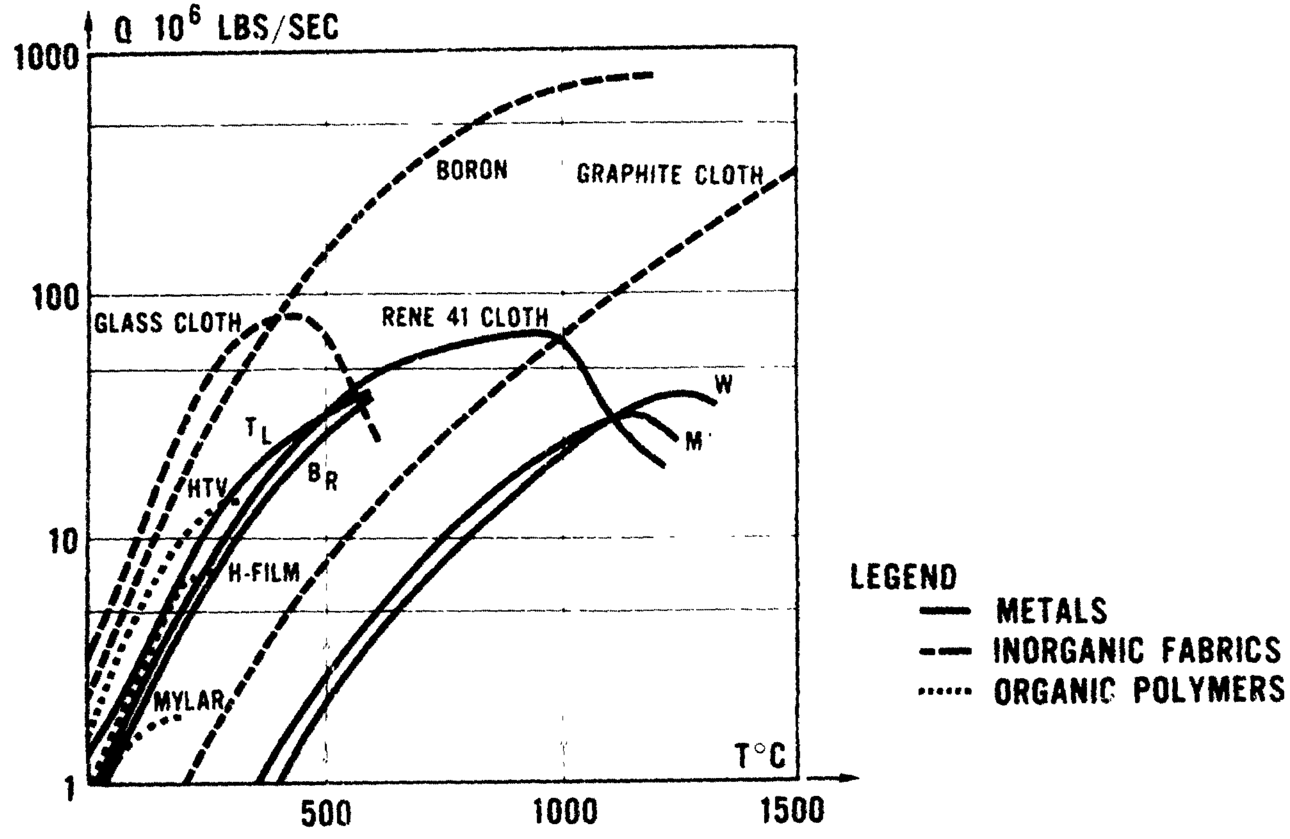


Figure 13 - Materials Comparison for High Temperature Use

booms. The sail and inflated booms of the glider are constructed of silicone-elastomer coated glass cloth. An additional ply of an aluminized-Mylar meteoroid sensor material is used on the upper and lower surface of the sail. With a reentry angle of trim of 42^{o}, and a velocity of 5500 ft/second, the structure will be heated to a peak temperature of 650-700^{o}F. Figure 14 shows the vehicle under test in the wind tunnel.

Other projects are under study using paragliders for spacecraft recovery. Most of these will involve applications at subsonic speeds.

Manned Orbital Laboratories - The NASA has made a number of studies of orbiting manned spacecraft for long periods for various purposes. Recent studies (Reference 5) have shown that one of the most important missions for long duration earth-orbiting vehicles is that of a space laboratory. Two classes of laboratories are being investigated. One is of compact form for a small crew and operates at zero g. The other class encompasses larger types and provides artificial gravity.

While it is evident that the size of spacecraft will vary with the duration of the mission and with the number of men, the provision for artificial gravity will in itself affect size. Since gravity can be simulated by rotation, many concepts lending themselves to stable rotation have been investigated, the most notable being the toroidal configurations. Radii of rotation and rates of rotation are also partly dictated by a consideration of biological effects. Large dimensions inevitably result from these parameters, and these in turn dictate expandable structures for launching a complete system in one package.

Figure 15 is an example of one type of orbiting laboratory of sufficient size to require a "packaged" launch configuration. The arrangement shown is a three module radial configuration accommodating a 24-man crew. The radial modules would be folded to provide a compact launch configuration. In space, the modules would deploy to a radius of 75 feet and rotate at 3-4 rpm. With a maximum centrifugal force at the outer decks of about 1/4 or 1/3 g, the station would provide a means of obtaining variable gravity down to zero at the hub. A multiple-walled rigid structure is being studied for the radial modules with sufficient protection against meteoroids for the long mission. An advantage of this design is the minimization of sealing required in going from the folded to the deployed configuration. Space laboratories involving pliant materials are also under study. This research includes investigations of various filamentary-elastomer wall constructions.

Lunar Bases -- Research is being directed towards solving the hypothetical mission for manned bases on the moon. Concepts for

Figure 14 - Meteoroid Paraglider

Figure 15 - Orbiting Space Station

these follow the sequence of first, a minimum initial shelter; second a temporary base; and third, permanent shelters.

The initial shelter could be carried aboard an excursion vehicle. In this case, it might have to fit within the dimensional constraints of the vehicle itself. This indicates that the shelter structure and perhaps its contents might be expandable.

The temporary base could be preassembled in modules and ferried to the extra-terrestrial location to be joined with others to provide enough volume and shelter for semi-permanent operations. Whether or not the modules would be expandable depends upon the particular design chosen. Permanent bases might be constructed on the planet using planetary materials. However, depending on material availability and construction difficulties, some components may have to be transported from Earth. Studies of bases are continuing.

Other Expandable Applications - Many of the equipment items utilized in space missions will be of an expandable nature. This is true insofar as the materials chosen and the structure into which they are fabricated have all of the characteristics of expandables even though their use may not involve a completely expandable operation. One such example is the space suit. In essence, a space suit must provide many of the elements of protection and convenience required from the space vehicle itself. Its thermal control, leakage rate, even its strength, must be adequate to provide the reliable enclosure required. If the often-pictured function of man-operation outside the spacecraft is considered, then meteoroid protection and greater radiation protection may also have to be added to the suit's function. This will be a function of exposure time.

GENERAL RESEARCH

Research on the design and operational parameters for various expandable space applications just reviewed reveal many common structural and material problem areas. Specific design of components and vehicles, and the short range research associated therewith, will be investigated as part of the particular application development when such development occurs. The long-range general research projects are chosen to solve the problems in structures and materials used in expandable applications. In fact, research in this field may often provide technology useful to all space vehicle structures.

The nature of spacecraft structural design in expandables as well as other types dictates major emphasis on thin films, foils, filaments and fibres as structural elements. Indeed, the extremely high strengths associated with small diameter filaments causes them

to be important for all structures. Parametric studies of systems often result in defining sizes and shapes which call for the very extreme in practical dimensions. Therefore, the combination of these requirements with the available material forms often results in structures in which expandable characteristics are provided. This is already evident in the choice of materials for passive satellites or space antennas where over-all dimensions are large but wall thicknesses are extremely small. The conventional sheet-metal-bulkhead approach to solving any structural problem is bowing to the results of analyses which define structural needs in terms of mission related efficiency factors An example of this was already given in the discussion of reentry problems. In many cases non-metallic materials and pliant structures are defined. In cases where the structures are essentially rigid (before and after deployment), problems are confined to research in sealing methods and geometry studies.

A few areas of general advanced research may be cited. In order to identify the important problems in pliant structures in which the primary source of loading is internal pressure, a study was conducted using weight as an index of merit. The parameters covered in the analysis were applicable to practically every type of expandable structure involving stressed membranes.

A few examples are presented to show the form of presenting information from these studies. All of the parameters studied are reviewed in Reference 10. Figure 16 shows the effect of shape and method of construction on structural weight. Low values of C are desirable. As shown in the figure, isotensoid construction is the only system wherein weight is independent of shape effects. However, not all shapes are adaptable to this construction. Studies of shapes which permit optimum isotensoid arrangements are reviewed in Reference 11.

Figure 17 shows the changing relative importance of strength-weight ratio. It can be seen that, as structural weight is decreased by the use of more efficient materials and construction methods, that other system weights do not necessarily change accordingly.

These studies have also emphasized two other important items. The first is that, although isotensoid structures are theoretically a less efficient method of providing a pressure retaining shell or surface, they permit use of materials and methods which can be furnished in high strength forms. Forms such as glass filaments more than compensate for the lack of structural efficiency. Conversely, the need is shown for development of high strength quasi-isotropic material forms, such as high strength films or foils. It is contemplated that as the potential use for such products increases, the development will be provided.

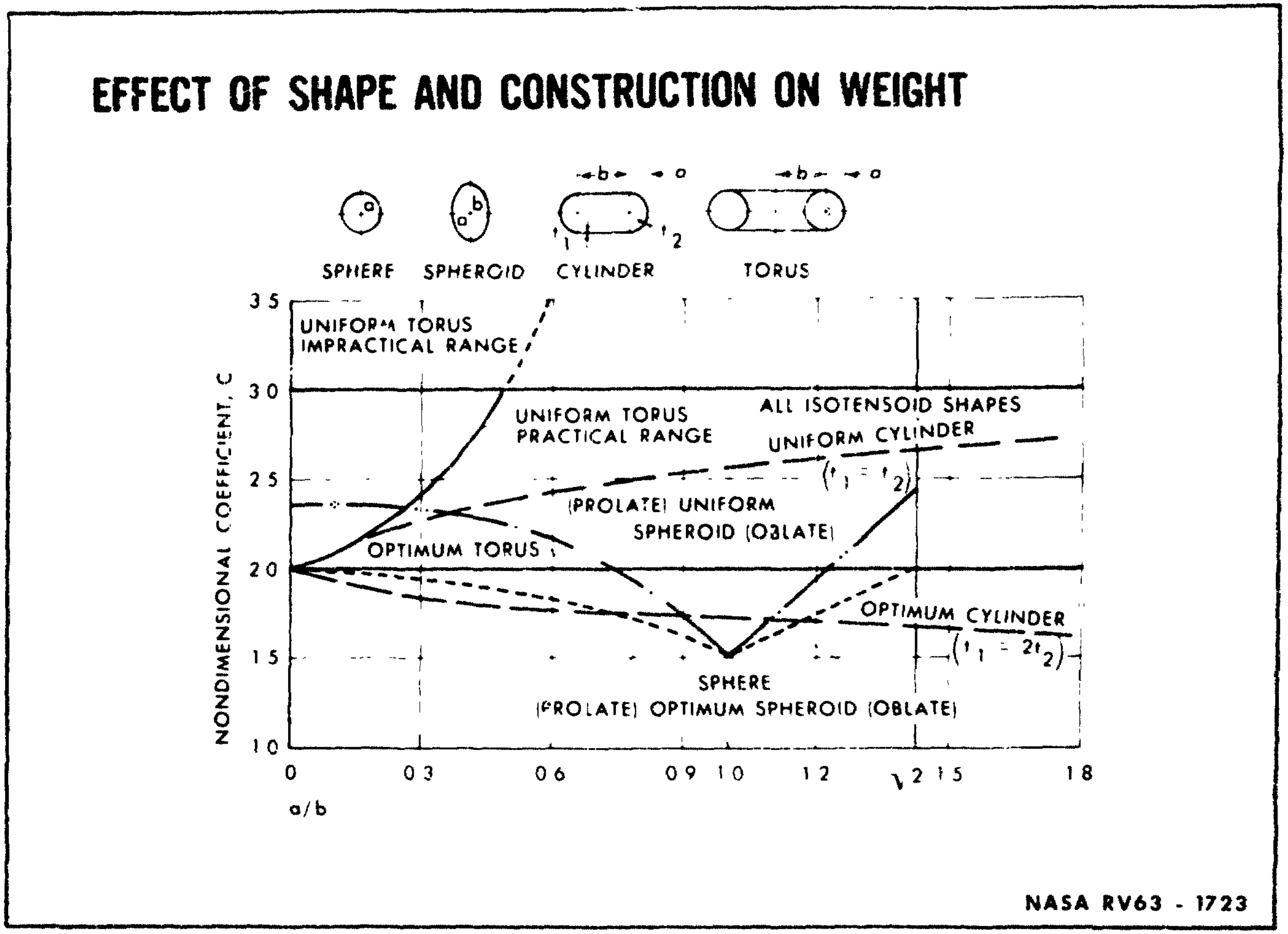
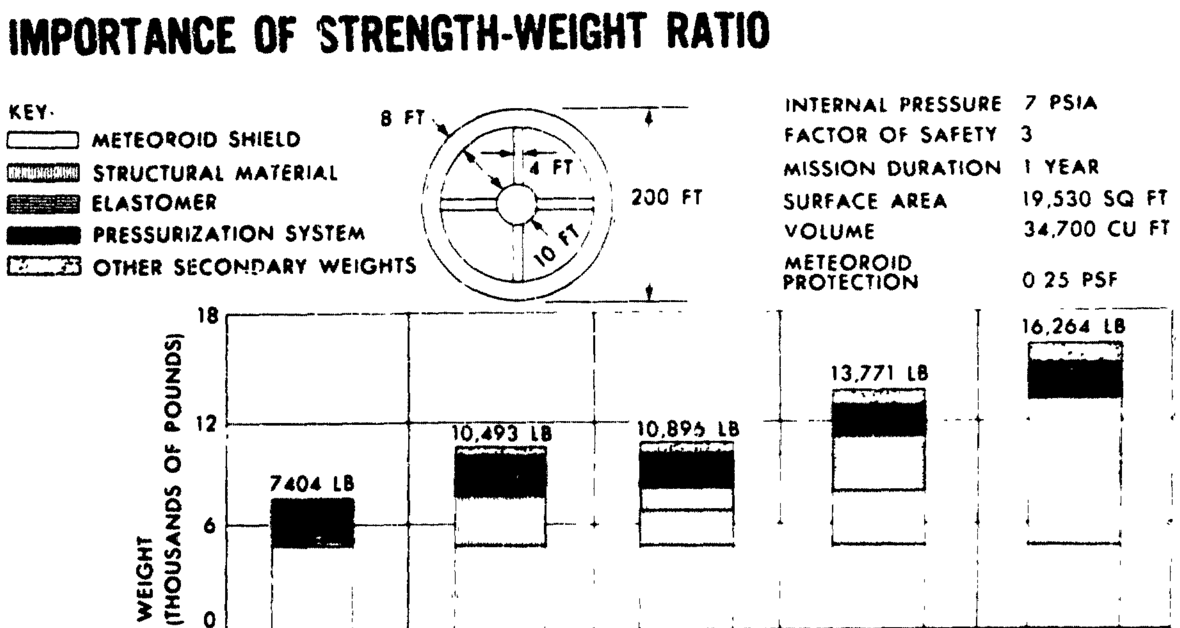


Figure 16 - Effect of Shape and Construction on Weight



MATERIAL	COATED WHISKERS	HYPOTHETICAL FILM	COATED FILAMENTS	SQUARE-WOVEN COATED CLOTH	MYLAR FILM
CONSTRUCTION	ISOTENSOID	UNIFORM SHEET	ISOTENSOID	UNIFORM THICKNESS	UNIFORM SHEET
STRENGTH-WEIGHT RATIO, k_1	12×10^6 IN	0.85×10^6 IN	1.70×10^6 IN	0.85×10^6 IN	0.311×10^6 IN

NASA RV63 - 1724

Figure 17 - Importance of Strength-Weight Ratio

A common problem among most expandable structures is the packaging of these systems for transport to the point of deployment. Some of these have already been discussed. Folding of expandables for packaging is susceptible to analytical treatment. Such studies were made wherein limiting equations were established for various shapes. A detailed discussion of the theories developed is presented in References 12 and 13. Figure 18 shows a typical example of a torus folded in accordance with the theory.

Since some expandable structures are both deployed and stabilized by utilizing centrifugal force, studies were made to develop governing equations for axisymmetric, rotating, filamentary structures. It was found that not only are resulting shapes predictable, but that the analytical methods also apply to non-spinning, externally loaded shells as well. Analyses and results are reported in References 14 and 15.

Other methods of expansion have been considered. One promising system utilizes the elastic energy in the structure to expand the configuration. An application of this principle was shown in the discussion of antenna design. It is obvious that this principle lends itself to many applications. General studies presently underway include determinations of design parameters, exploration of various elastic systems, and selection of optimum arrangements. Figure 19 is a picture of a laboratory model of a cylinder, which utilizes elastic expansion

It is obvious that primary emphasis in advanced research has been devoted to the development of technology in filamentary and non-metallic structures for the cases cited previously. However, NASA has also conducted general studies of metallic forms of expandable structures composed of rigid elements. A summary of these studies is contained in Reference 16.

An example of the application of a rigid element expandable structure for advanced research is shown in Figure 20. This is a satellite designed to provide meteoroid penetration data during earth orbit.

In the pursuit of understanding of pliant materials and expandable structures in space, it is becoming evident that many valuable techniques in both analysis and development are being provided which will benefit not only expandables but all space structures. The potential in non-metallic structures is especially large. Even if it were not, the need for knowing more concerning non-metallic construction applied to space vehicles will continue. It is almost impossible to design systems utilizing nothing but metals. It is certainly not practical. On this basis, greater technical effort must be devoted to material developments and design techniques which lend themselves to expandable and to non-metallic structures. Among these, the more important seem to be:

Figure 18 - Folded Torus

Figure 19 - Elastic Expandable System

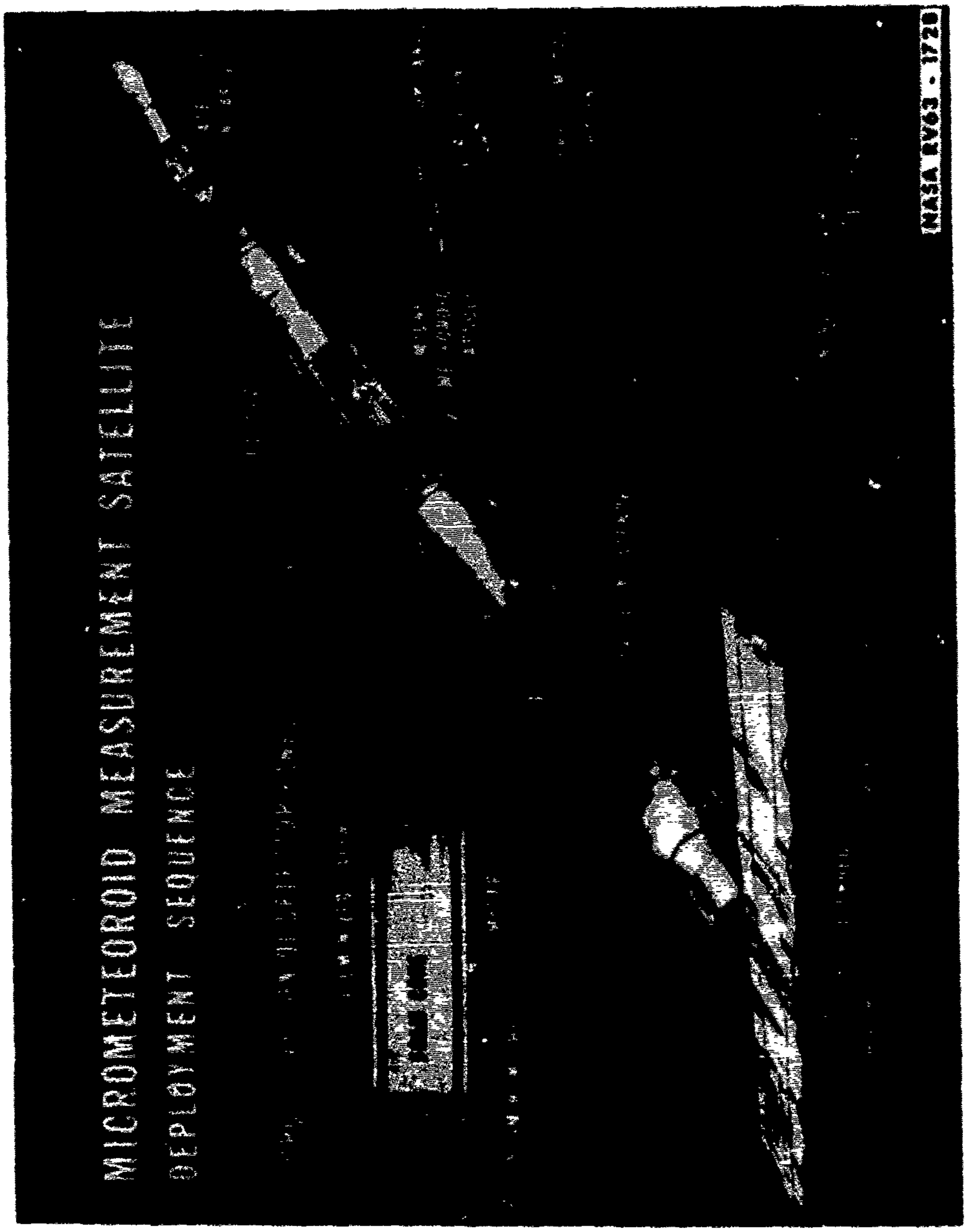


Figure 20 - Micrometeoroid Measurement Satellite

1. Increased development and understanding of filaments as structural elements.

2. Improved films of higher strength and modulus of elasticity.

3. Use of parametric analysis as a first step in choosing a structural approach to particular requirements.

4. Greater emphasis on the design of materials for particular structural applications (e.g., composites).

CONCLUSIONS

Expandable structures in space applications have been chosen because studies of mission requirements have shown expandable concepts as the optimum choice. However, many problems will continue to be manifest in both structural design and applications of materials in these structures. More work is needed on non-metallics in filament form and films. The use of parametric studies in choosing structural solutions and proper materials should be increased. The number of individuals and groups with specialized knowledge of design techniques and familiarity with the materials available will probably continue to represent a minor part of those engaged in spacecraft design. Even fewer are those who possess the combination of ingenuity, imagination, and technical ability to investigate really new approaches. It is conceivable, then, that there will always be opposition to departing from the rigid metal systems on the basis of tradition. These considerations represent the largest problem area in the future of not only expandable structures but for other advanced concepts as well.

FIGURES

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SELF-DEPLOYING SPACE STATION

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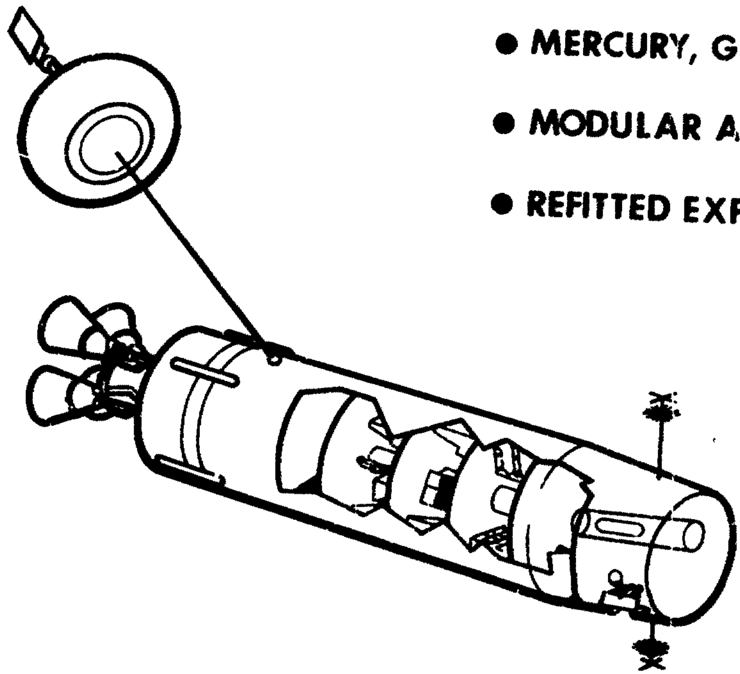
The objective of this paper is to introduce some overall systems constraints into the use of inflatable materials or expandable structures in spacecraft design. In particular, the discussion is related to how the use of these concepts might influence or be influenced by a large manned space station.

There are a number of general requirements which can be established for any space station that is to be developed for conducting experimental or test activities. First of all, the economic factors associated with the operation and support of an extensive experimental program indicate a superior advantage to a system which has the capability of conducting several activities simultaneously. Also in the interest of reducing program cost, it is desirable that any vehicle which is launched into orbit should have the ability to remain in operation for an extended period of time, perhaps as long as one to three years. The need for an extended lifetime is further substantiated by the requirement that many of the foreseen experiments have a test duration of several months. In order to accomplish the basic vehicle functions, it is estimated that at least two crew members should be available for duty on a continuous basis. Thus, a brief investigation of work/rest cycles indicates that the crew size will be no less than four to six people. To accommodate the experimental activities and provide a suitable living and working environment over the extended periods of time, a large volume and payload capacity is required for the space station. While it has not been finally determined that man can or cannot survive for extended periods of time, it is believed that a space station should simultaneously be able to provide a variable-gravity and a zero-gravity environment to accommodate foreseeable experimental activities. To minimize the development cost associated with any space station, compatibility with existing or planned personnel transport vehicles (Apollo, Gemini, and X-20) and launch vehicles is essential.

Zero gravity space stations have been examined in considerable detail. Actually the existing manned spacecraft, i.e., Mercury, Gemini, and Apollo, are small zero-gravity space stations; however, in their present configurations they do not have the large volume and payload capacity required for extended missions. Studies are now being conducted to examine the feasibility of increasing the available volume of the Apollo by the addition of a modular laboratory. Expended booster stages, modified in orbit as space stations, also appeared attractive at one time. The North American Aviation study of the use of the Saturn S-II tankage for this purpose (see Figure 1) is typical of many designs which have been considered. The primary problem associated with this approach is the requirement for the in-orbit installation and checkout of functional systems and subsystems. The extensive instrumentation and associated equipment required for this task appears



- MERCURY, GEMINI, APOLLO
- MODULAR ADDITION TO APOLLO
- REFITTED EXPENDED BOOSTER STAGES

Figure 1. Zero-Gravity Stations

to offset any advantages which might be achieved with such a concept.

Vehicles that have been designed specifically as zero-gravity space stations have a limited number of factors that influence the configuration selection. Since compatibility with the launch vehicle is the predominant criterion, these space stations generally have an elongated cylindrical shape.

Inflatable space station concepts are particularly desirable because of their apparent ability to be packaged in a more compact launch configuration than is possible with all-rigid concepts. This approach is particularly attractive whenever a large volume in-orbit is desired, but where available boosters have extremely limited payload weight and geometric constraints. Obviously, pre-launch installation of equipment is not practical if the packaging efficiency is to be maintained. Figure 2 illustrates a typical 24-foot diameter "C-Annulus" inflatable space station which was built under contract to the NASA Langley Research Center. Normally, equipment will be installed within a central ring and later set up inside the laboratory by the crew members. An alternate approach would be to install the equipment around the interior ring of the space station; however, when the station is rotating in orbit, the crewman's feet will be at the periphery of the station and all of the equipment will be a couple of feet above his head. One of the most serious problems associated with the use of an inflatable material is its unsatisfactory resistance to meteoroid penetration. Materials that have relatively high packaging efficiencies provide practically no meteoroid protection; the space station atmosphere would soon escape, and subsystem equipment would be seriously damaged. If the material is made resistant to meteoroid penetration by the use of multi-wall construction, the packaging efficiency rapidly approaches that of conventional rigid structure.

Many suggestions have been made for space station concepts involving the launch of several individual components and their assembly in orbit. These are generally very large space stations and engineers have often worked out a sequence of operations for the assembly of these vehicles. Figure 3 depicts a concept that was recently patented by the Lockheed Aircraft Corporation. However, the assembly of all the components in orbit is a much more difficult process than simply launching a number of pieces into a general location in space and collecting all of these at a common point by means of a shuttle vehicle. The problems of mechanical attachments, fluid connections, electrical line connections and checkout of the integrated system are only a few which tend to make such a concept not practicable before the post-1975 time period.

The concept of a self-deploying space station incorporates most of the characteristics that are highly desirable for an early operational space station. As was discussed previously, the use of the space station as a general-purpose laboratory will require a large volume to accommodate equipment and to provide living and working space for crew members. The incorporation of artificial gravity capability, required at least for experimental purposes, leads to need for large-dimensional configurations which can be slowly rotated. The initial configuration of a self-deploying space station which was studied by NAA under NASA contract is illustrated in Figure 4. It is composed of six rigid modules joined by inflatable

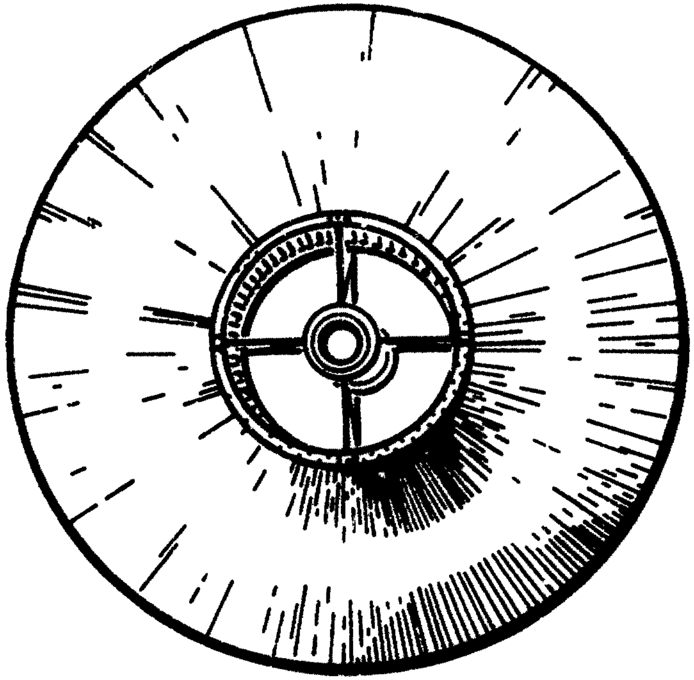


Figure 2. Inflatable Concepts

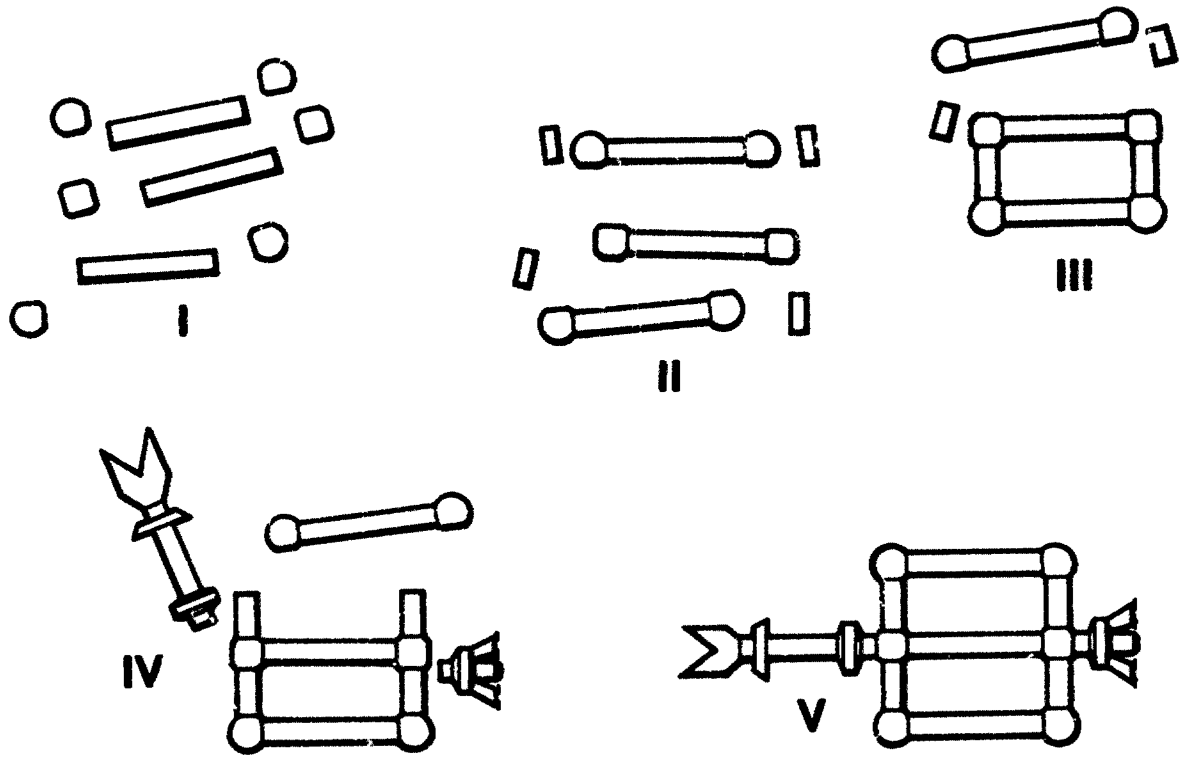


Figure 3. Assembly Sequence

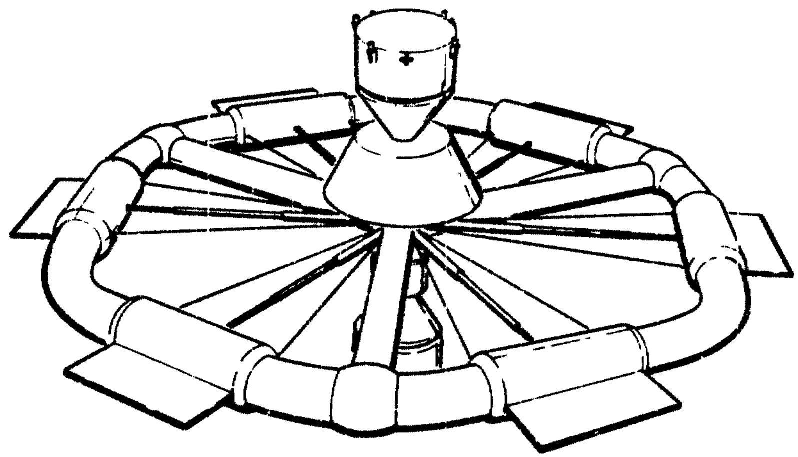


Figure 4 nitial SDSS Configuration

material and arranged in a toroidal shape which is rotated to provide artificial gravity. Three spoke joined the torus to a central hub which was used as a docking platform for two personnel transport vehicles (each comprised of the Apollo command module and service module). The three spokes were to be constructed of inflatable material which could be folded, thus enabling the compact launch configuration shown in Figure 5 to be achieved. The six rigid modules, which have all systems and equipment installed and checked out prior to launch, are clustered around the central hub section. One Apollo vehicle, which could be launched with the space station, serves as the means of escape for three crew members in the event of booster malfunction and launch abort. When the orbit has been successfully established, the folded space station is deployed automatically by means of hydraulic actuators, electrically-driven screw jacks, or some other mechanical device. Struts connecting the hub with each of the six rigid cylindrical sections are used to assure uniform deployment from the launch configuration to the orbital configuration. The orbital configuration, with the non-rigid sections inflated, results in a space station with a 100-foot diameter, although the diameter could be treated as a variable. One end of the hub has been enlarged to house a small zero-gravity laboratory. The deployment sequence for a model of this space station is shown in Figure 5.

While the diameter of the initial configuration was sized by the payload capability of the then-planned (circa 1961) Saturn C-2 booster, it became apparent early in the investigation that the rotational effects on the crew members might be intolerable. In Figure 6, a number of the parameters affecting a crewman's tolerance to the conditions aboard a rotating space station have been plotted in a manner such that a "comfort zone" is defined. The bounds of this comfort zone are determined by four parameters - the percent change in gravity between the man's head and feet; the space station angular velocity; the tangential velocity; and the radial acceleration. It is believed that the change in gravity or acceleration between the man's head and feet should not exceed 15% in order to prevent impairment of blood circulation and to reduce the discomfort felt when the man changes the relative position of his head and feet by bending over or lying down. The recommended upper limit on the space station angular velocity is approximately 3 rpm to minimize the effects of Coriolis acceleration. Experiments with the USN School of Aviation Medicine rotating room have indicated that normal head movements are tolerable if the angular velocity does not exceed 3 to 4 rpm. The tangential velocity of the space station should be no less than 20 fps in order to minimize the change in gravity the man will experience when he walks along the rim of the station. The upper limit on radial acceleration, or gravity, was set at 0.5 g to reduce fatigue; the lower limit is somewhat arbitrary, it being the minimum acceleration under which normal functions (e.g., locomotion) can be expected to be performed in a near-normal fashion.

It can be seen from Figure 6 that a space station radius of 50 feet does not fall within the defined comfort zone. With the increased payload capability of the Saturn V booster, a space station diameter of up to 75 feet appeared practical. As will be shown later, the weight of this vehicle is compatible with the Saturn V payload capability in a low-altitude orbit. Consequently, a radius of 75 feet was selected for subsequent space station designs.

The space station will be continuously subjected to meteoroid bombardment during its lifetime in orbit. Since the inflatable material which was

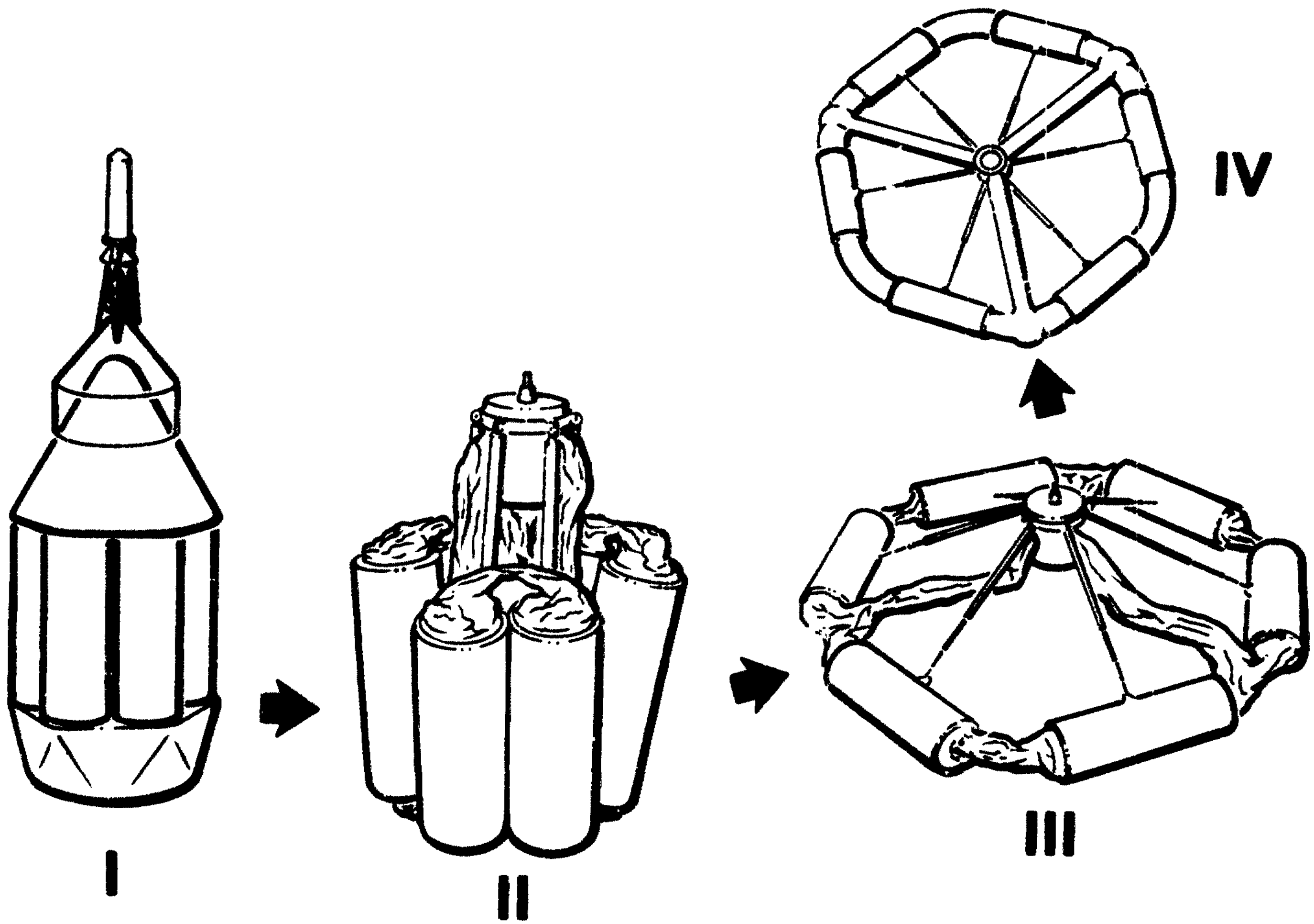


Figure 5. Deployment Sequence

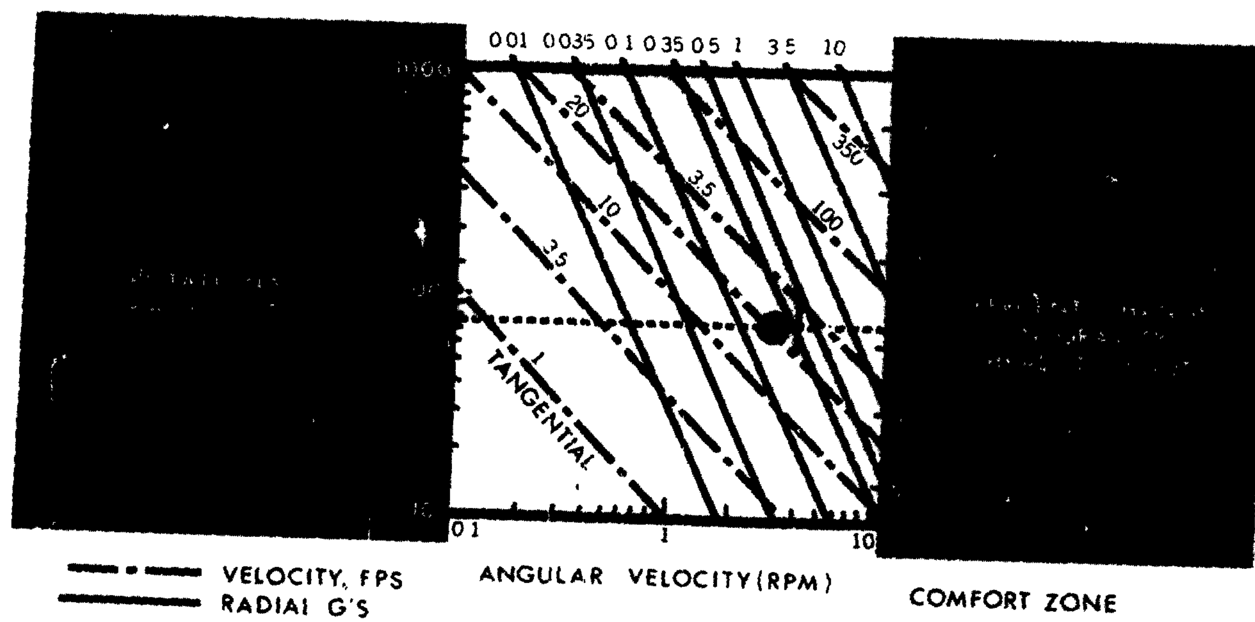


Figure 6. Rotational Considerations

contemplated for use in the design has very poor penetration resistance, its use was minimized on all designs. Wherever inflatable material was chosen for design use, it was to be lined with penetration-resistant panels by the crew members after the station has been deployed. The meteoroid flux distribution that was assumed for this study is based on the somewhat conservative modified-Whipple 1961 estimate illustrated in Figure 7. As indicated in the figure, the majority of the particles that will be encountered in space are very small and cause only a gradual erosion of the exterior surface of the vehicle. It is estimated that in a year's time, the amount of aluminum removed by micrometeoroid sputtering is of the order of 10^{-6} to 10^{-5} inches. Of particular concern are those particles that are sufficiently large to cause a penetration of the structure. Empirical data derived from hypervelocity particle-impact tests at the Ames Research Center were used in conjunction with this flux to determine penetration effects. Efficiency factors for single sheet and multiple sheet structure have also been developed. Using these efficiency factors with the flux and penetration equation, it is possible to determine the meteoroid resistance of various types of structure. Figure 8 illustrates the variation in weight of four typical wall structures as a function of the number of meteoroid penetrations (penetration by particles of all sizes) per year over the entire vehicle surface. It can be seen that a multi-wall aluminum structure is much more efficient than a single-wall structure; however, the efficiency does not appear to increase beyond a three-wall structure. For a 75,000-pound structure in a vehicle of the size under consideration, it appears practical to restrain the number of penetrations by meteoroid particles to the order of one per year.

With the advent of the large boosters (i.e., Saturn V), the necessity for examining inflatable space station concepts vanishes. Human factors requirements have indicated the necessity of increasing the diameter of the rotating station to approximately 150 feet and meteoroid penetration problems have indicated the obvious desirability of minimizing the use of inflatable material. Thus the configuration evolution indicated in Figure 8 preceded. The second configuration shown in the figure has constrained the use of inflatable material between the modules to the minimum length necessary to fold the vehicle in the launch configuration. Further analyses, supported by the construction of a number of crude models, indicated that if the appropriate hinging techniques were employed, the inflatable material between the modules could be completely eliminated. The only exposed inflatable material being used in the third configuration in the figure is in the three spokes connecting the hub to the modules.

Further refinements of the 150-foot space station configuration indicated that even the inflatable material in the spokes could be eliminated if the spoke were constructed as a telescoping tube with a pinned joint at the hub and a pinned rotating joint at the module. This configuration, illustrated in Figure 9, is composed of six modules which are 75 feet long and 10 feet in diameter. The three telescoping spokes have a maximum external diameter of 5 feet. This design has evolved to provide the maximum flexibility of operation to the crew members performing the various functional activities aboard the station. Crew members have access to all compartments at all times, yet a design technique has been employed to give maximum safety under all circumstances. Each of the modules, as well as the hub, has its own power, environmental control, and life support systems. In addition, a network of airlocks has been utilized so that any major section of the space station can be isolated in event of a serious failure. For example, if a large meteoroid penetration occurs in a module, an airlock at each end of

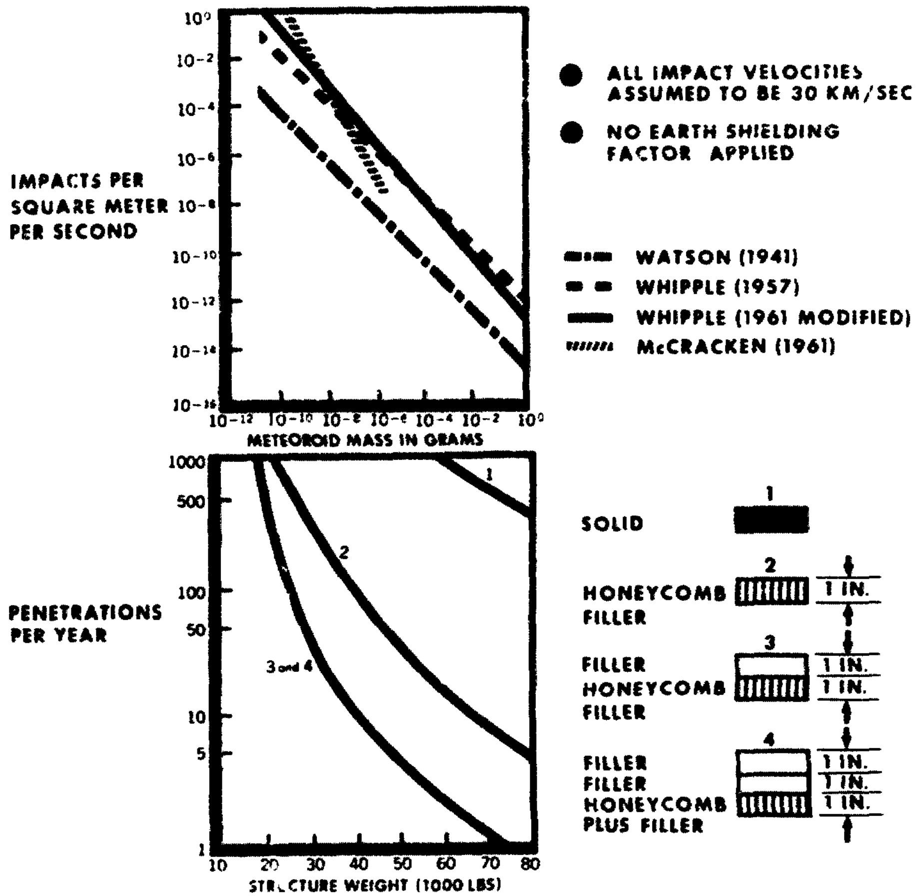


Figure 7. Meteoroid Considerations

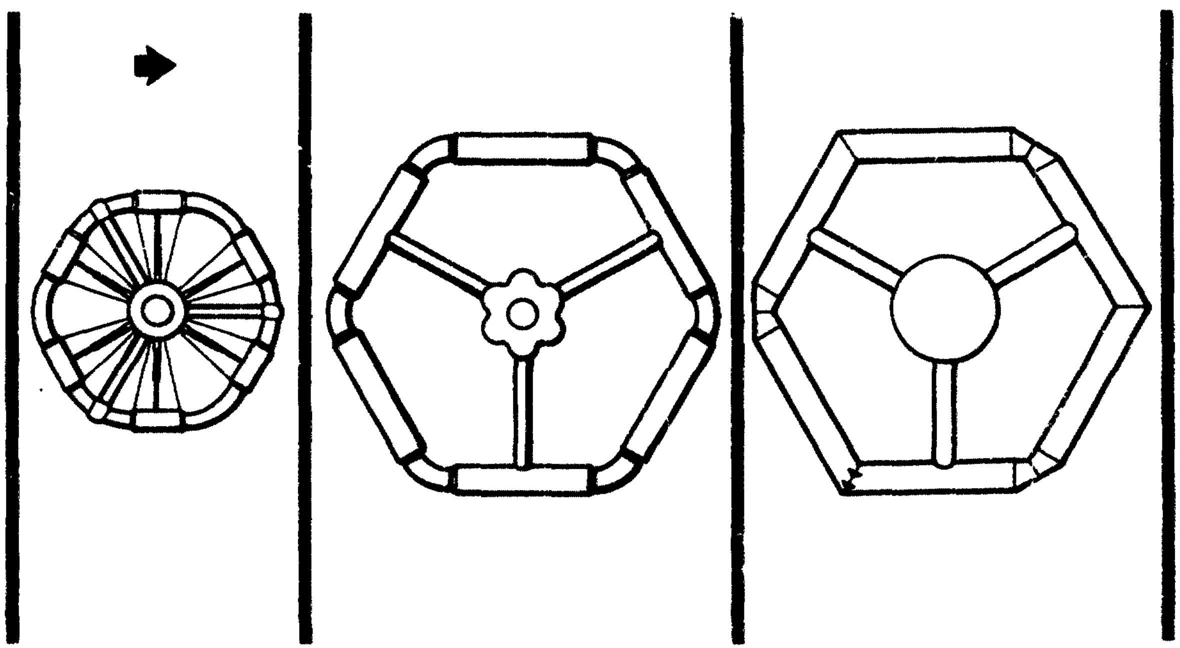


Figure 8 Configuration Evolution

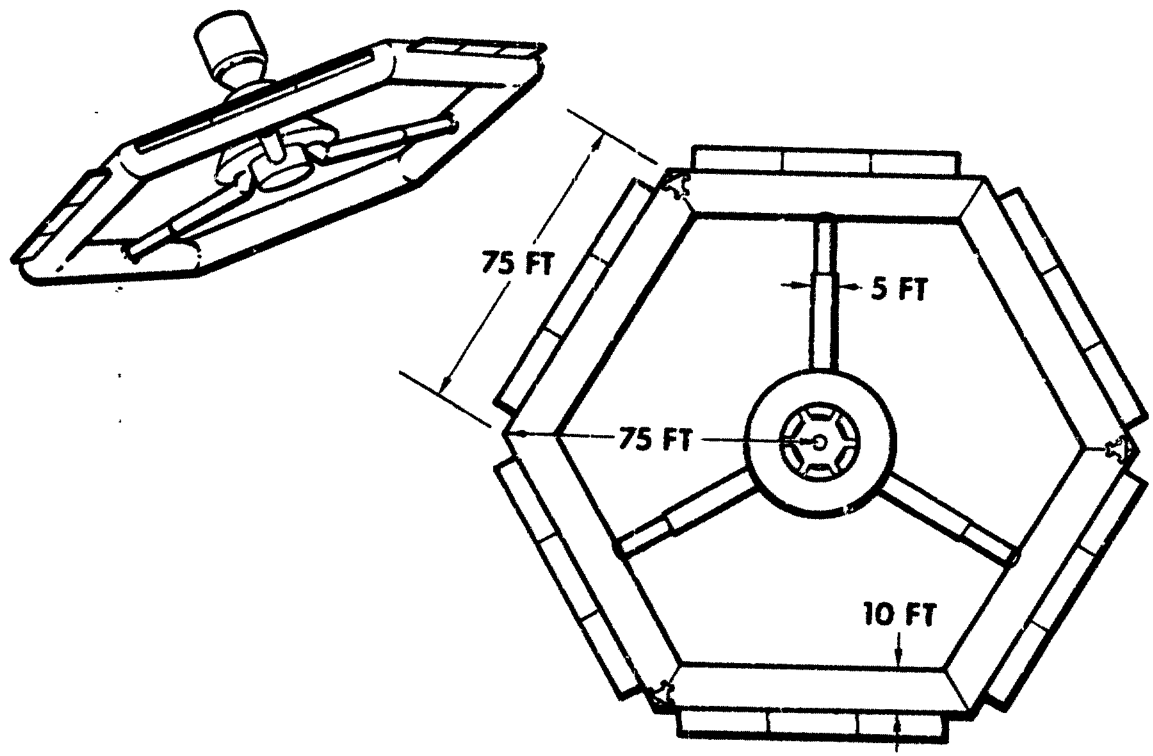


Figure 9. Typical SDSS Design

the module will prevent loss of the remaining atmosphere in the station and, because major system functions are duplicated, normal space station operations can be continued. Crewmen may enter a damaged module through the airlock to conduct whatever repairs are necessary to restore the module to its original state of operation. One of the requirements originally established for a space station was that it be capable of simultaneously providing both a zero-gravity and artificial-gravity environment. This station has a 3200-cubic-feet, zero-gravity laboratory within the hub; the remaining volume (approximately 45,000 cubic feet) can be used for living and working quarters and conducting experiments that are independent of gravity level.

The structure of the space station—the modules, the spokes and the hub—is composed of three layers of aluminum sheet as indicated in Figure 10. The outer layer is 0.020 inches thick and the inner two layers are 0.038 inches thick. The inner two layers are separated by a one-inch thickness of aluminum honeycomb core, the cells of which have been filled with polyurethane foam. The outer two aluminum skins are separated by a one-inch layer of polyurethane which has been bonded to each skin. Several longerons designed to resist launch bending loads in the structure divided the cylindrical sections (modules and spokes) into 60-degree segments. The figure illustrates a typical way in which the manufacture of the wall structure might be accomplished.

The longerons, fastened in a jig to maintain the proper location, are welded to the preformed inner sheets of the structure by the use of the "skate" welding technique developed at North American. Polyurethane foam will be deposited in the honeycomb core, and, after the honeycomb has been cut to shape, a small amount (0.010 inches) of the foam will be removed from each cell so that the bonding agent can firmly adhere to the honeycomb. The assembly will then be placed in an autoclave to cure. After curing, the second layer of aluminum skin will be welded to the longeron and the assembly again placed in an autoclave to cure the bond between the sheet and the honeycomb. The layer of polyurethane foam is now bonded in place on the inner sheet of aluminum, then the outer sheet (the meteoroid bumper) is bonded to the foam and riveted to the longeron.

The design of all the seals that are used in the space station was selected on the basis of minimum initial leakage and the ability of the crew members to perform subsequent operations that would result in a permanent zero-leakage joint. Since the space station will not be folded after initial deployment, all of the joints in the structure which are necessary for folding the station for launch can be made permanent by the use of bonding or welding techniques. The seal concept which has been employed in the telescoping spokes. It makes use of two circumferential lip seal glands which are held against a smooth service by the internal pressure of the space station atmosphere. This technique automatically forms a low-leakage seal after deployment has been completed. A manual adjustment has been incorporated so that crew members may tighten the seal to further diminish leakage and, if desired, the joint may be welded shut to ensure that the leakage rate is reduced to essentially zero. It should also be noted that this same adjustment can be used to compensate for any changes in spoke length due to temperature changes or separate misalignments in manufacture of final assembly. While it is possible to conceive many complicated seal mechanisms for such a device, it is believed that simplicity and ease of subsequent provisions for a permanent seal are the key to minimum leakage in vehicle joints such as those described.

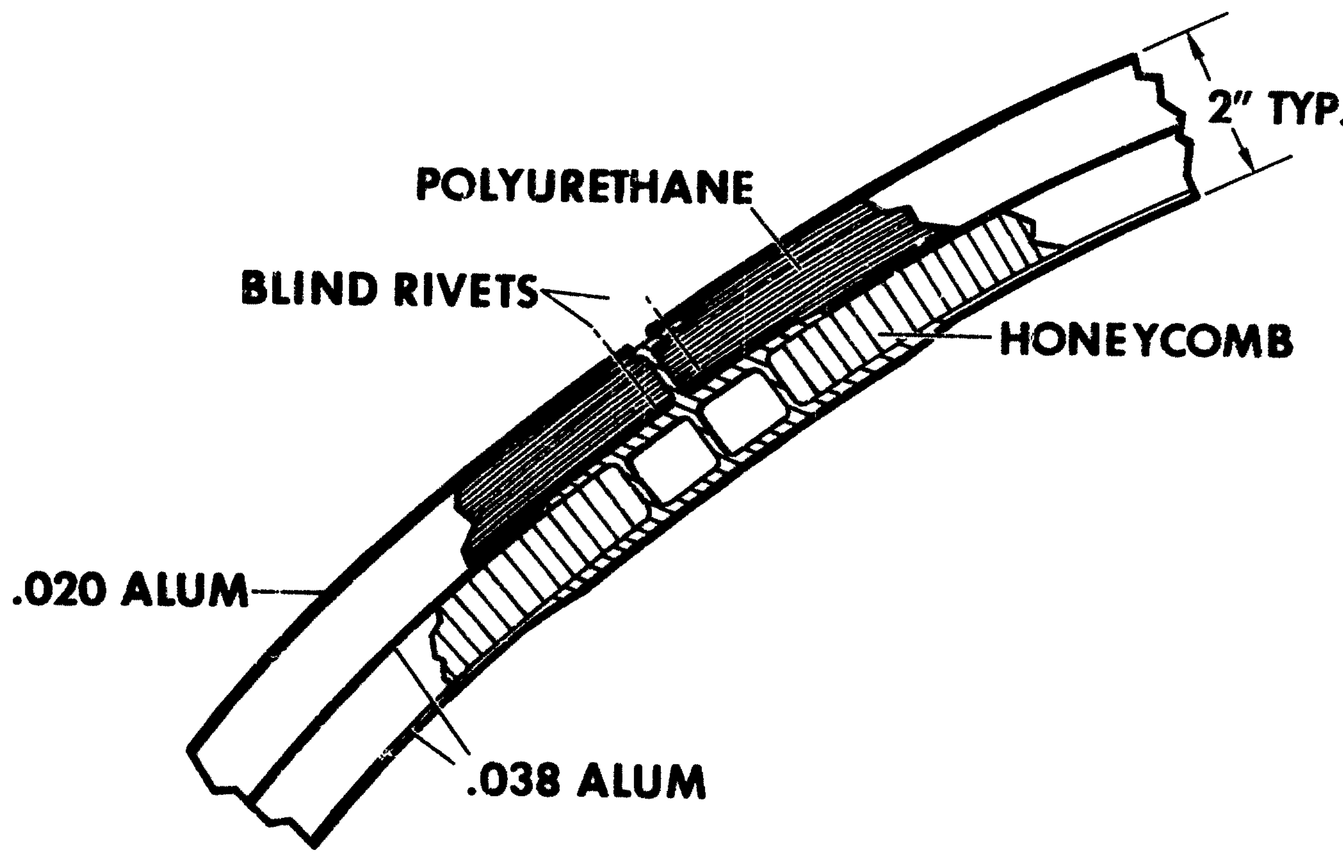


Figure 10. Typical Wall Structure

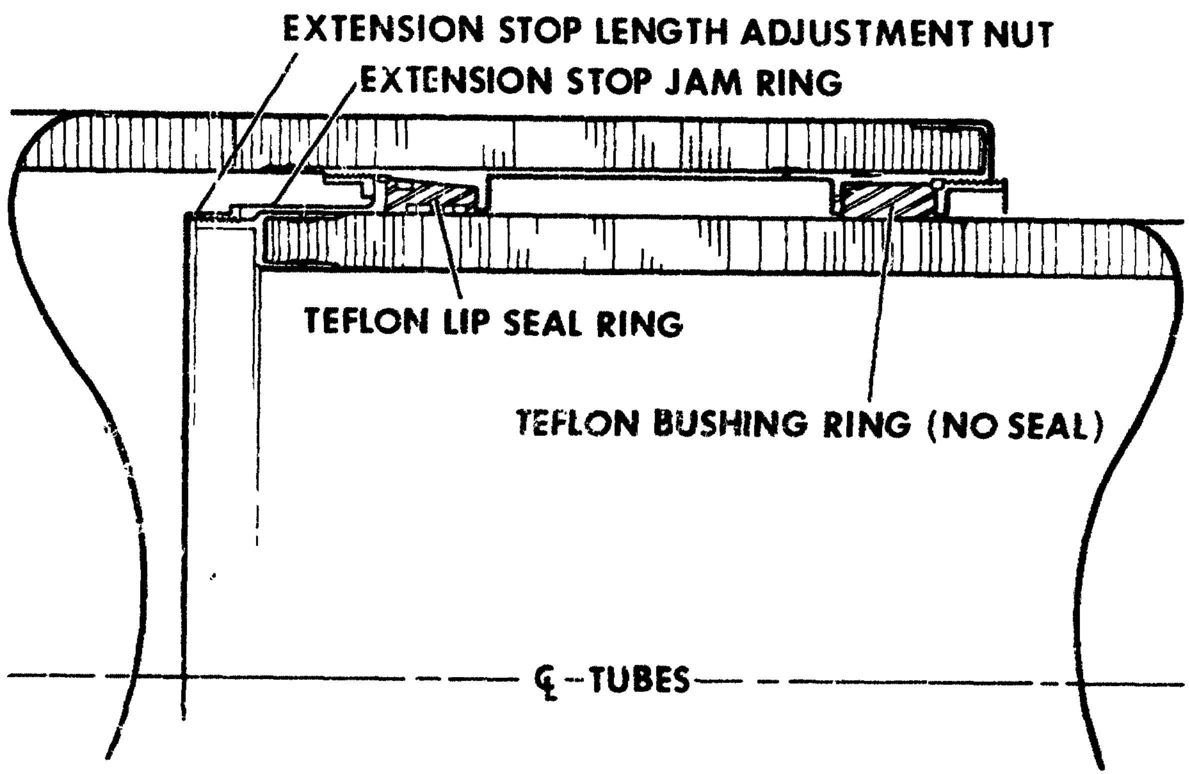


Figure 11 Telescoping Spoke Joint

The details of other major structural elements, as well as associated manufacturing techniques of the space station, have been worked out in a manner similar to that previously described for the wall structure and for the telescoping spoke joint. Because of the limitations imposed on the presentation of this paper, however, no further details will be shown.

Because of the numerous questions concerning the ability of such a station to be satisfactorily deployed without binding of the joints, a one-tenth-scale self-deploying model of the station was constructed under contract to the NASA Langley Research Center. The model is powered by self-contained electric screw jack actuators. Two actuators are located at each module joint and three are installed in the hub to control the spoke motion. It is recognized that this actuation mechanism does not duplicate the actuation concept for the large space station; however, the principal of coordinated deployment is identical. Although the motions of the 15 actuators are not precisely coordinated, it was found that the model would deploy satisfactorily. It was discovered that limited lag in the motion of any one actuator could be tolerated without binding, although it is obvious that a failure of an actuator could result in serious damage to or possible destruction of the model. The model has pointed out the need for either a coordinated deployment mechanism or a station concept where coordinated deployment is not required.

To overcome the necessity for coordinated deployment of the space station, the concept illustrated in Figure 12 was derived. In this configuration, the telescoping joints in the spokes have been eliminated and the spokes now meet the rim of the space station at the intersection of two modules rather than at the midpoint of the modules. This particular concept illustrates the use of rotating seals at the joint between the spoke and the modules; however, butt seals would be equally appropriate with a slightly different design.

The deployment of this space station occurs in two distinct phases as indicated in Figure 13. In neither phase are the motions of the sets of modules coordinated and the second phase of the deployment is not dependent upon all the modules completing the first phase of the deployment. In the first phase, three sets of modules and spokes (a set consisting of two modules and one spoke) deploy to form the letter "Y". The second phase of deployment involves the rotation of adjacent modules about a center line passing through the ends of adjacent spokes to form the complete hexagonal configuration. If a malfunction occurs during the deployment sequence, crew members may enter the station and manually complete the deployment sequence prior to the operational activation of the station. As in the previously described concept, the seals at the joints are designed in such a way that welding or bonding techniques may be employed by the crew members to form a permanent seal.

A summary of the weight of the previously described space station concept is illustrated in Figure 14. The structure of the vehicle, which includes mechanical devices necessary for deployment, weighs 75,450 lbs. The weights of the functional subsystems--power, environmental control, communications, stabilization and control, and life support--are indicated on the chart. With a payload allowance of 21,000 lbs. for scientific instrumentation, test equipment, etc., the space station launch weight is only 138,600 lbs. This is, of course, well within the payload capabilities of the Saturn V launch vehicle (S-IC and S-II stages).

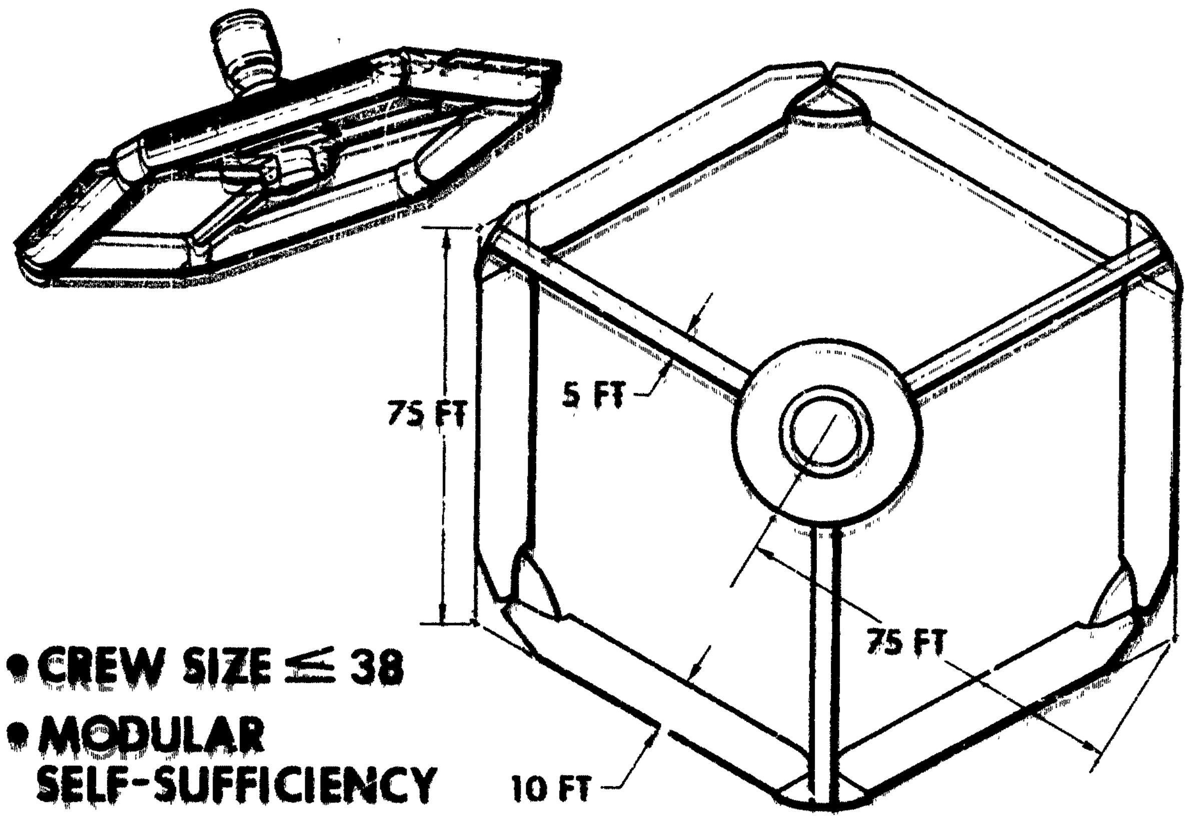


Figure 12. Current SDSS Design

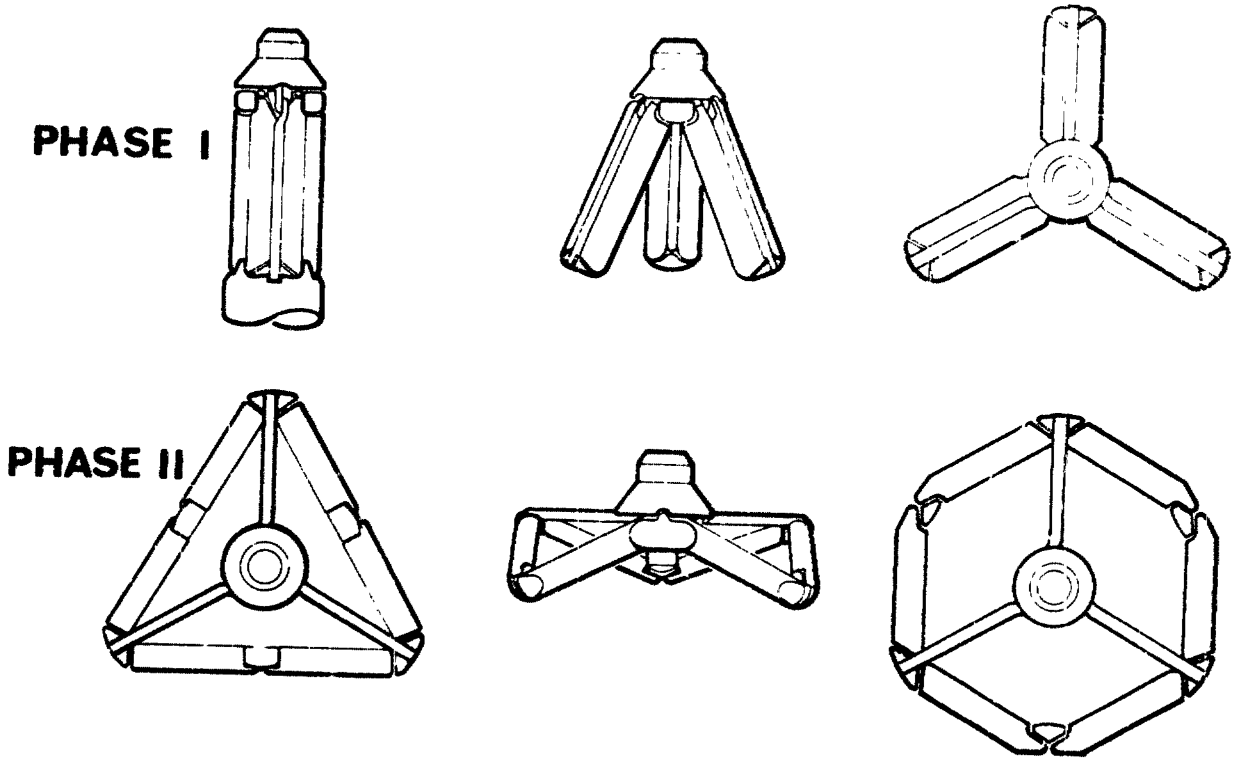


Figure 13. Deployment Concept

75,150	STRUCTURE
11,280	POWER
22,976	ENVIRONMENTAL CONTROL
1,262	COMMUNICATIONS & DATA DISPLAY
3,400	STABILIZATION & CONTROL
3,532	PERSONNEL ACCOMMODATIONS
21,000	PAYLOAD (TEST EQUIP, SCI INSTR)
SPACE STATION WEIGHT 138,600 LBS	
10,000	INTERSTAGE
LAUNCH WEIGHT 148,600 LBS	

Figure 14 Weight Summary

In general, expandable structures do not represent an optimum solution, per se; they always represent a compromise in comparison with a completely rigid structure. However, volumetric or other dimensional constraints imposed by certain operational considerations have at times been so severe that only the use of expandable structures permit the accomplishment of the mission

Both space station concepts discussed herein—one involving two-mode deployment and the other utilizing telescoping spokes—are expandable structures in that the orbital configuration must be "packaged" or folded to be compatible with the aerodynamic restraints imposed on the launch vehicle.

These studies have fairly conclusively demonstrated that systems which require the use of expandable structures can be subjected to creative design approaches to eliminate many of the problems normally associated with the expandable structures.

ABSTRACT

DEVELOPMENT OF AN INFLATED PARAGLIDER FOR MANNED ORBITAL ESCAPE

J. F. Keville and F. Warren

Space-General Corporation
El Monte, California

The problem of developing a material system for use in an inflatable structure which would be capable of hypersonically re-entering the earth's atmosphere was undertaken by Space-General Corporation under sponsorship of the Manufacturing Technology Division of the Air Force Materials Laboratory. The specific problem was to develop a material which would possess sufficient flexibility to allow folding into a compact package while being strong enough to withstand the required internal pressure and the aerodynamic loads during flight. Additionally, the material system would have to withstand extreme heating effects during re-entry while maintaining low permeability so as to retain the inflation gas.

The approach has been towards the use of currently available materials with design criteria based on studies of the related trajectory and thermodynamic environments. The general approach has been to use elastomeric materials with low ablation losses and yet a relatively modest effective ablation temperature. Since the low thermal conductivity of the elastomers and the ablative char would tend to minimize the transfer of heat to the reinforcing material, the design allows the use of available reinforcing wire cloth. A critical aspect of the problem is solved by the development of a joining technique for the wire cloth to achieve the desired shape and load handling ability. This is followed by the application of an elastomeric coating to the reinforcement to provide the necessary heat protection and gas barrier.

Results of testing performed to date are disclosed and include evaluation of various elastomers by plasmajet tests and torch tests, evaluation of various methods of joining the wire cloth, coating the wire cloth with the elastomer, and tests of the composite material.

The significance of the results of this program is that materials available today can be combined to form a composite system suitable for use in an inflatable structure which is capable of re-entering the earth's atmosphere from orbit. Such materials may be applied to a variety of special missions usilizing the Rogallo wing paraglider.

DEVELOPMENT OF AN INFLATED PARAGLIDER FOR MANNED ORBITAL ESCAPE

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The importance of human survival has received considerable emphasis in our space program. As a means of escape from an orbiting space station, an inflated glider-type vehicle has been proposed. A collapsible, flexible wing vehicle, which is particularly aerodynamically stabile, was conceived by Francis Rogallo nearly twenty years ago. The principal application of this vehicle has been as a subsonic, relatively low altitude, glider with an underslung payload.

Under a contract awarded by the Manufacturing Technology Division of the Air Force Materials Laboratory, Space-General Corporation has undertaken the initial development of a completely flexible, inflated paraglider which will be capable of returning a man from orbit by re-entering the earth's atmosphere at hypersonic velocity and decelerating to effect a subsonic controlled landing on earth. The program has been given the title "Project FIRST," which stands for "Fabrication of Inflatable Re-entry Structures for Test."

A photograph of a model of the proposed paraglider is shown in Figure 1. The vehicle consists of an inflated body made up of three inflated tapered booms attached to a common toroidal apex. Attached between the center boom or keel and each leading edge boom is a thin, flexible wing membrane which assumes an approximately semi-conical shape during flight.

The vehicle configuration, which has been confirmed by theoretical analysis and extension of data obtained in wind tunnel testing, employs booms tapered from thirty-two inches in diameter at the forward end to sixteen inches in diameter at the aft tip with an overall vehicle length of approximately twenty-three feet.

Figure 1. Re-entry Paraglider

Figure 2 Deployment of Escape Vehicle

The vehicle is packaged similar to a life raft and attached to the space station such that a man can enter the crew compartment section of the paraglider from within the space station. The vehicle is then ejected away from the space station and subsequently inflated. Having been aligned at the proper attitude, the vehicle is retro fired from orbit and re-enters the sensible atmosphere at 26,000 ft/sec., or about Mach 30. An artist's concept of the deployment of the paraglider is shown in Figure 2.

The development program for this vehicle includes the design, fabrication, and test of large scale components. In order to proceed with design, it was necessary to first determine the critical environment so that materials of construction could be chosen. One of the project criteria was that "available" materials must be used. This was interpreted to mean that materials research leading to new alloys or formulations was outside the scope of the project, although currently available materials might be modified in adapting them for this application.

The trajectory and thermodynamic studies, materials evaluation, and structural analysis and design have been grouped into Phase I of this project. This phase has been completed recently. Phase II will include process development, tool design and fabrication, and actual manufacturing of the test components.

These components will consist of both full-scale and sub-scale booms and apexes which will be tested in Phase III under simulated load and thermal conditions.

Evaluation of the pressure distribution on the sail membrane and keel at hypersonic velocities was made using an approximation to the Newtonian theory. With the spanwise pressure distribution established, the resulting aerodynamic coefficients during hypersonic re-entry were determined as functions as angle of attack as shown in Figure 3. In the range of hypersonic velocities considered the variation of these coefficients with Mach number is small. The parametric study of heating and ablation rates indicated that minimum heating occurred near maximum lift coefficient. As a result, a vehicle angle of attack of 70^{o}, which corresponds to an L/D of 0.5 was selected. After evaluation of a number of trajectories using an IBM 7090 digital computer, an aerothermal corridor corresponding to minimum re-entry heating was found to occur at a re-entry angle of -1^{o}. The trajectory characteristics are shown in Figure 4. The vehicle re-enters the sensible atmosphere where significant heating first appears at 420,000 feet altitude and within about 1800 seconds has descended to 130,000 feet where it is approaching subsonic velocity. The re-entry maneuver incorporates only one well established skip and glide above 330,000 feet and the resultant energy dissipation is reflected in a decrease in velocity while still at relatively high altitude. Not only are the heating rates maintained at relatively low values as will be shown later, but the dynamic

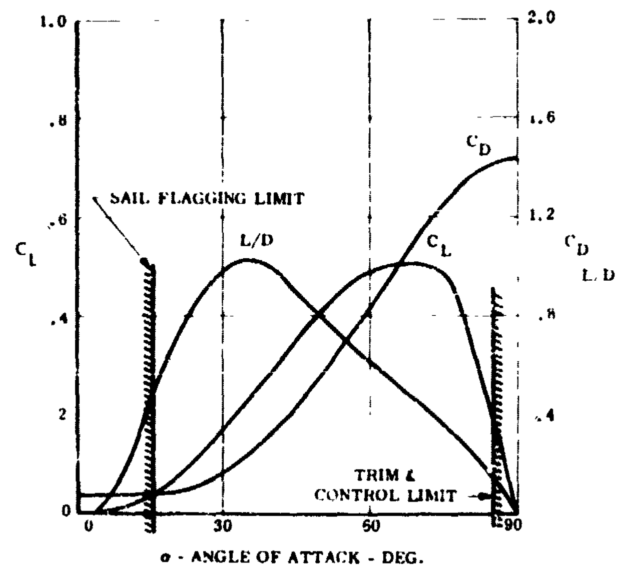


Figure 3 Aerodynamic Data

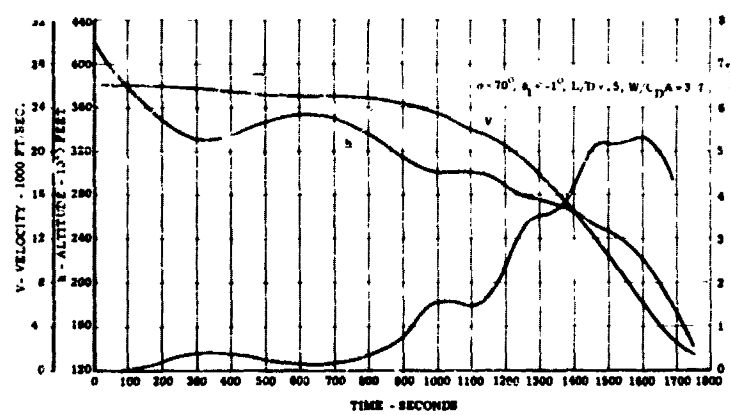


Figure 4. Trajectory Characteristics

pressure reaches a maximum value of only 5.3 lbs./sq. ft., and this is in the latter part of the supersonic regime. At about this time, the maximum deceleration, which is tangential to flight path, peaks out at less than 1.4 g.

Thermodynamic studies were performed using the IBM 7090 computer to evaluate the nature of the temperature distribution at the apex and along and around the leading edge booms. Due to decrease in diameter at the tapered booms, the aerodynamic heat flux actually increases from forward to aft of the vehicle. At the aft end of the boom the heat flux is actually greater than at the stagnation point on the apex. Figure 5 shows the thermal history experienced by the vehicle. The temperature curve showing a maximum or

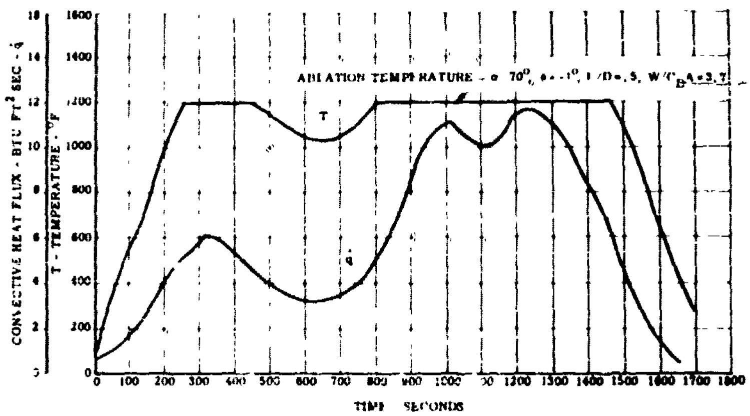


Figure 5 Thermodynamic Data

ablation temperature of 1200°F is based on the actual thermal properties of the coating material as determined in plasma-jet experiments. The program in its early stages was iterative in nature since assumed materials properties had to be used to determine the materials temperatures which, in turn, were used to guide the selection of final materials.

Materials evaluation started with a survey of literature of available materials. Since suppliers had very little knowledge of high temperature of ablation characteristics of their materials, it was necessary to conduct an extensive materials testing program. Candidate materials were evaluated from the standpoint of their performance in the flight environment and their fabricability.

Due to the high flexibility and crease resistant requirements imposed by the need for compact packaging, elastomeric materials appeared to be the best candidates for the coating materials. Previous work by our company had disclosed that the silicone rubber materials were very satisfactory as ablators in lining of rocket nozzles and insulation against base heating.

A series of oxy-acetylene torch tests were conducted on standard silicone rubber formulations and special formulations made up in our laboratory which included the use of ceramic frits, micro-ballons, etc. As a result of these torch tests, certain silicone elastomers were selected for plasma-jet testing. Specimens of the candidate materials were exposed to heat fluxes ranging from 1.5 to 15 Btu/sec., sq. ft. in an eighty KW air-arc plasma-jet. Typical results are illustrated in Figure 6. Non-silicone elastomers showed

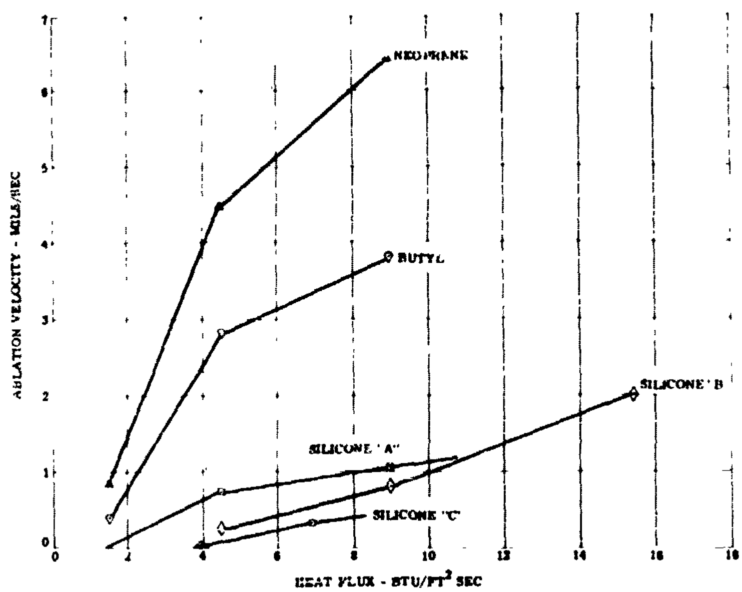


Figure 6. Plasma-jet Test Results

very poor performance by comparison. Recalling that the maximum heat flux shown in Figure 5 was less than twelve Btu/sq. ft., sec., average ablation velocity during the flight is well under 1 mil per second for a silicone rubber

typical of that finally selected for coating this vehicle. Although the differences between torch and plasma-jet testing were expected to cause considerable discrepancy in the results, it was interesting to observe the materials performed relatively the same and even the ablation velocities were similar for the particular conditions utilized.

Permeability tests were run using air and nitrogen and it was found that thin coatings of the silicone materials would withstand the design gas pressure with negligible gas diffusion when reinforced by lightweight stainless steel tensil bolt cloth for time periods and temperatures well in excess of those to be experienced in the actual flight. Calculations indicate that less than 0.25 lbs of gas would be lost while the total pressurizing and attitude control gas carried on board is about 27 pounds.

Considerable testing using a tensile test machine was conducted on one- and two-ply coated and uncoated specimens of metal cloth. Tensile testing was also the primary method of defining the quality and efficiency of the various joining methods considered for both glass and metal cloth reinforcing fabrics. Since tensile machine testing cannot create a bi-axial load condition, small inflated cylinders were constructed which not only gave considerable insight into the joint performance and characteristics of incipient buckling during bending, but also provided experience in fabrication techniques.

The criteria for structural design and resulting design loads which were computed are shown in Table 1. Since no rigid structural reinforcements

Table 1. Structural Criteria and Design Loads

STRUCTURAL CRITERIA

1. Thin membrane structure - Pressure stabilized
2. No buckling at 120% limit.
3. Hypersonic air loads are critical

DESIGN LOADS

1.	Internal working pressure	11 PSI
2.	Maximum hoop tensile load	176 LB/IN
3.	Maximum longitudinal load	159 LB/IN
4.	Minimum longitudinal load	17 LB/IN
5.	Keel shear load	15 LB/IN

were to be used in the vehicle, pressure stabilization is required to prevent vehicle distortion under aerodynamic and inertial loads. It was assumed that

sufficient pressurization would be required so that buckling would not occur at 120% of limit loads while the critical design loads would be encountered during the hypersonic portion of the flight. Although gust loading during subsonic flight may momentarily exceed the more sustained loads to which the vehicle is exposed in a hypersonic flight, a lower temperature, greater maneuverability, and less critical adherence to a prescribed trajectory in subsonic flight are expected to alleviate this condition.

The calculated internal pressure of 11 psi produces a maximum hoop tensile load of 176 lbs/inch. Although the nominal longitudinal load in a cylinder would be half of this value, the maximum longitudinal load due to air loads bending moments is 159 lbs/inch while the minimum is 17 lbs/inch This latter value illustrates the effect of pressure stabilizing in excess of design loads.

The maximum hypersonic air loads in the horizontal plane are shown in Figure 7. It will be noted that these loads created by air impact and wing

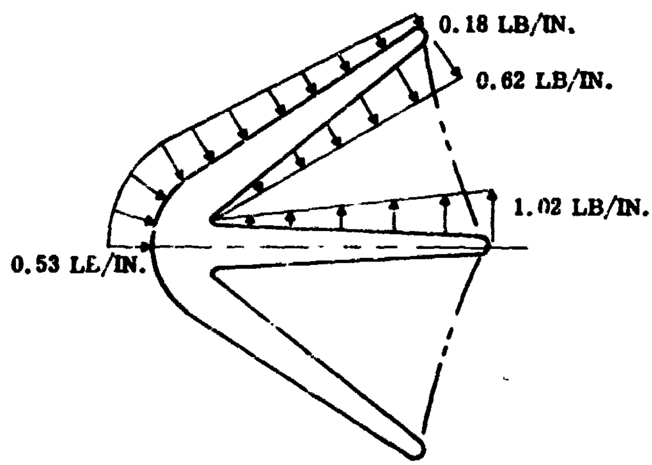


Figure 7. Maximum Hypersonic Airloads (Horizontal)

membrane tension are very low as are those shown in Figure 8 for the vertical air load components and inertial loads due to crew compartment (in the keel boom) and structural mass distribution.

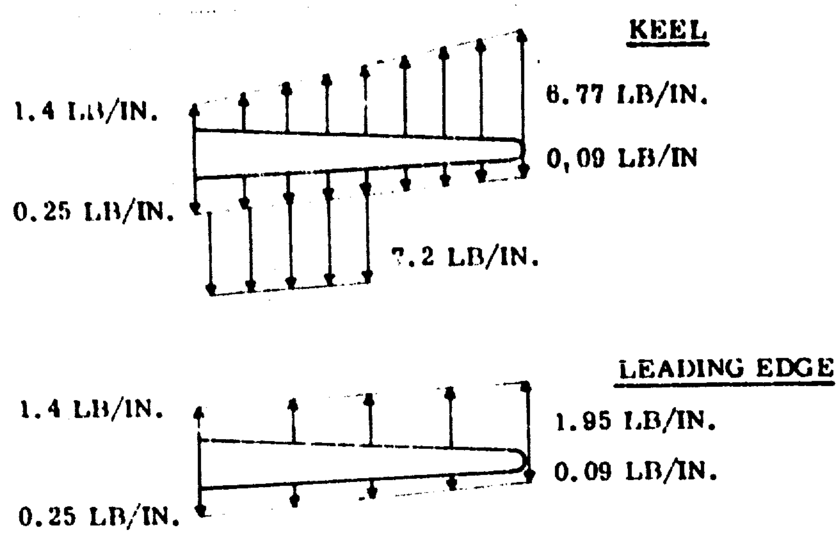


Figure 8. Maximum Hypersonic Airloads (Vertical)

Design of the material matrix is consistent with the temperature distribution illustrated in Figure 9. The upper curve illustrates the outer surface temperature and ablation previously discussed. The two lower curves show the temperature at the outer surface of the reinforcing fabric and the inner surface adjacent to the pressurizing gas. It will be noted that the reinforcing fabric at no time exceeds 1,000°F. This is desirable because from a weight economy and material availability standpoint relatively common materials could be considered for the reinforcing fabric.

The need for high flexibility with high crease recovery and high strength to weight ratio led to the decision to use ultra-fine filament stranded metal cloth. Laboratory testing of filaments as well as fabrics woven from 300 series stainless steel as well as chromium nickel alloys has led to the selection of an alloy containing 73% nickel and 20% chromium. This material is marketed commercially by one manufacturer as Chromel R. The fabric is woven in a 58 mesh 2 x 2 basket weave of yarns consisting of forty-nine 1 mil diameter filaments. The resulting fabric which is about 12 mils thick has a flexibility and texture of a similar textile fabric.

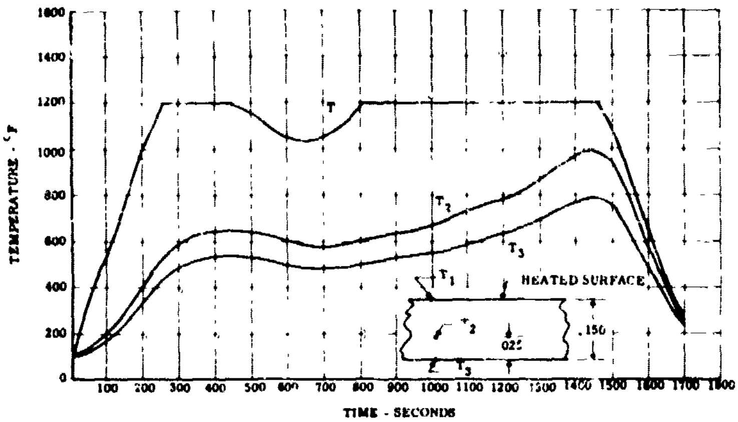


Figure 9. Design Data - Thermal

Two plys of this fabric will be used in the booms with the inner ply on a 45° bias while the yarns of the outer ply will run generally longitudinal and cross-wise on the boom. A cross section of this fabric matrix is illustrated in Figure 10. The wire cloth which will be thoroughly impregnated with silicone rubber will occupy approximately 25 mils of the total thickness. The

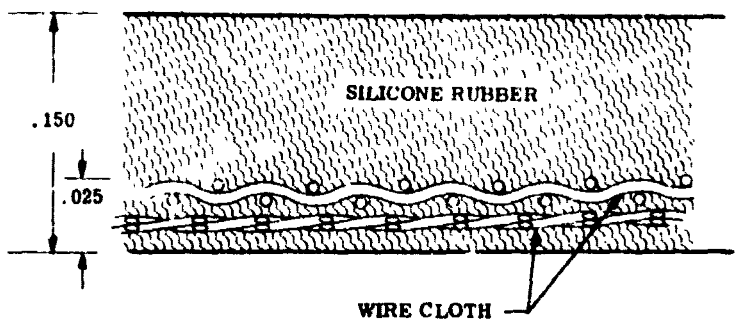


Figure 10. Material Composite

illustration shows the total thickness of silicone rubber required on the lower ablative surfaces at the aft end of the booms where 0.10 inches will ablate leaving 0.025 inches of silicone rubber still coated over the cloth.

The primary internal loads in the boom fabric are tension and shear. Since tests indicate that the silicone elastomer will carry a negligible portion of tension loads, hoop and longitudinal loads will be reacted by tension in the filaments in the outer metal cloth ply. As shown in Figure 11, the thin bonding

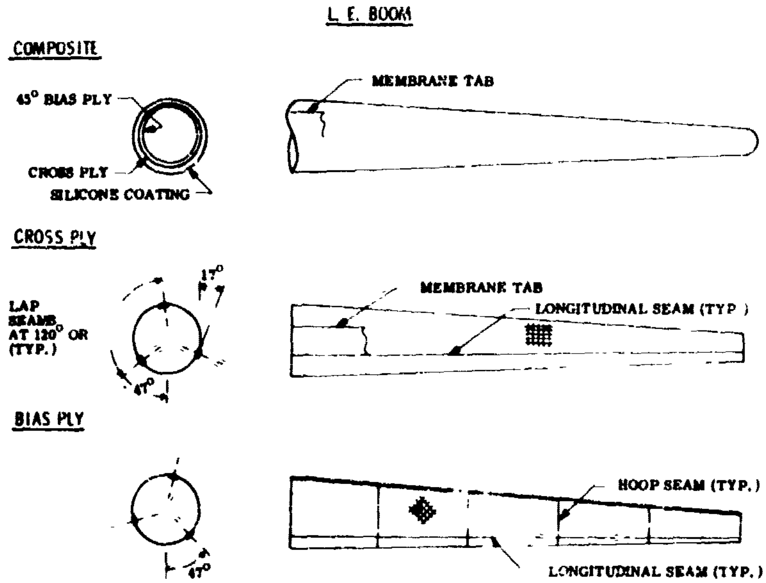


Figure 11. Boom Construction

film of silicone between the plys and application of seams joining both plys permit shear loads in the booms to be converted into longitudinal loads in the bias ply fibers. Longitudinal seams in both plys and hoop seams in the bias ply are dictated by maximum available fabric width.

The construction of the apex is considerably more complicated in that hoop loads are not carried continuously around the torus due to the intersection of the keel boom and bending moments on the booms must be reacted by the apex without buckling or gross distortion. As in the case of the booms, the apex will utilize a primary construction of two-ply in bias relationship. To react other peripheral loads in circumferential tension, the apex will then be "mummy" wrapped with strips of fabric as schematically shown in Figure 12.

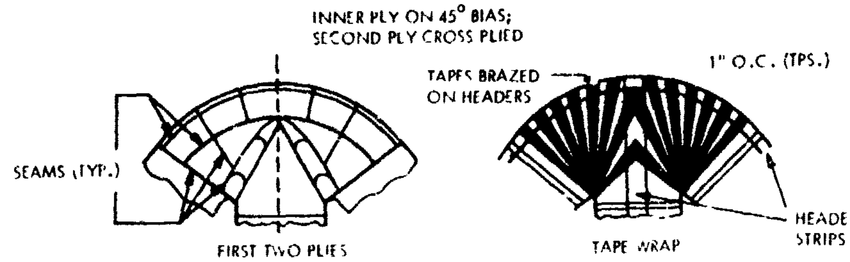


Figure 12 Apex Construction

Although the apex will be considerably less flexible than the booms, it is expected that it may be collapsed with a reasonable packing factor.

In general, the procedure for fabricating the vehicle components will consist of first cutting the metal cloth in the required pieces and carefully laying the pattern together with as many joints as possible being made on a flat surface The joining procedure is essentially a brazing technique performed in an inert atmosphere The primary method utilizes the 'Aeroray' infrared gun to concentrate heat from a quartz lamp on the joint with an eliptical reflector This heating device has the advantage of being portable, although all brazing must be done in a nitrogen or argon atmosphere which is accomplished by working with a plastic bag covering the entire work piece. Two alternate methods of preparing brazed joints are being developed. One of these utilizes standard resistance welding equipment to melt the brazing alloy but presently this method suffers from lack of portability although no inert atmosphere is needed. A third method which will require substantially more development involved the use of a small induction heating coil and, again, this method requires an inert atmosphere Joint efficiencies of 85 to 95% have been consistently produced.

The next step in fabricating a part is to shape the cloth patterns to a tool assuring conformance with the desired surface geometry of the vehicle. Final joints are made on the tool. The cloth shape is then either removed from the tool for solution impregnation by dipping in a solvent-cut silicone rubber mixture or impregnation is accomplished by brush coating. Subsequently, calendared sheets of silicone rubber approximately 0.025 inches thick supported by PVC film are laid in place to build up the total thickness as required in certain areas of the boom and apex. After the surface of the outer layer of silicone rubber is properly smoothed or skived as required, the entire assembly is vacuum bagged while still on the shaping tool and cured in an autoclave.

Testing of the components will include both room temperature and high temperature tests as well as corresponding tests conducted after simulated re-entry heating. These tests will include pressurization and permeability to ascertain gross leakage and diffusion rates as well as burst strength, determination of packaging factor, and loads at high temperatures after packaging, vibration surveys in both packaged and inflated conditions, and performance under design loads as well as determination of buckling loads. Since ablation characteristics cannot be adequately studied outside of the wind tunnel, attempts will be made only to heat the fabric in a peripheral temperature profile similar to that predicted for the re-entry condition. Since the auto-ignition temperature of silicone rubber formulations is 900 - 1000^{o}F, such high temperature tests including pressurization, permeability, and application of loads will have to be conducted in a nitrogen purged chamber. A schematic of a loads test at temperature is shown in Figure 13. The loading is small and easily simulated by the use of small weights in such a way that they do not interfere with the heating created by a bank of quartz lamps. Thermocouples on the wire cloth will be used to regulate the heat lamps to maintain the required temperature pattern. The boom will be properly reinforced where it is attached to the test fixture in order to prevent failure at the clamp due to load concentration.

In conclusion, it is hoped that distinct contributions will be made to the design and construction of expandable structures and that techniques developed in this program will either be directly usable or will point the way to further development.

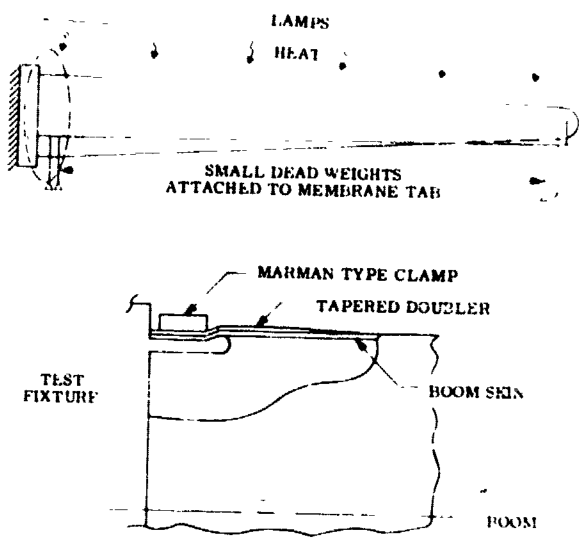


Figure 13. Boom Loads Test

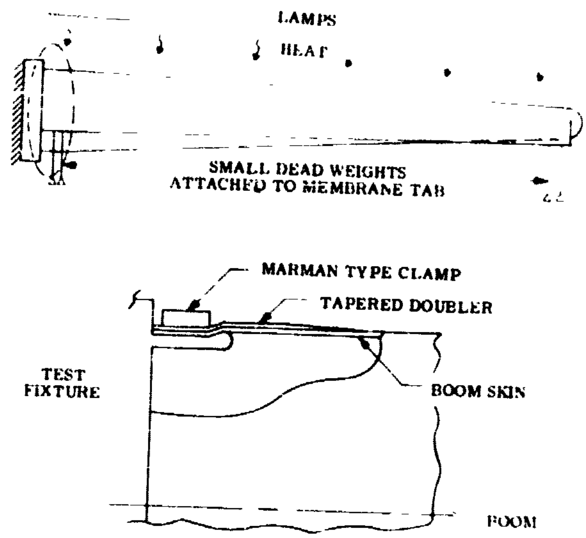


Figure 13. Boom Loads Test

MSFC INTEREST IN EXPANDABLE STRUCTURES

By

J. F. Blumrich
Chief, Structural Engineering Branch

MSFC's interest in expandable structures has resulted in quite detailed work that has been done over a number of years. Our experience, besides studies and related research work, includes actual development and design of flight hardware. We are glad, therefore, of the opportunity to present a brief summary of the most significant of our structures, and we want to emphasize the definite interest we have in this field - both combined with our eagerness to learn more from as many sources as we possibly can.

Before going into detail I should like to refer to Webster's Dictionary which explains the word "expand" with "increase in extent." This increase may - quite logically - occur along one axis only; it may also mean an increase in two directions, and it may naturally also involve three directions. This means to say that the field of expandable structures is wider than frequently considered. It includes, e.g., telescoping structures, structures that are deployed (like solar mirrors or flat panels of satellites, Regallo wing) balloons, and it could also include space stations and lunar shelters. These are examples for the major systems. However, we should also include here auxiliary structures which may be packaged on earth and put to proper shape in space or on the moon: Radiators, connections between shelters, even such items as tables, chairs, and probably a long list of other items.

Expandable structures are not limited to payload structures, of course. In our area of activities we find, however, that the majority of expandable structures does occur in the payload region. For this short discussion I will, therefore, consider major systems of payloads only.

The purpose of making a payload structure expandable is to store it, at least during the atmospheric flight of a vehicle, in such a way that either its length is substantially reduced or its diameter is kept at a desired size. This means that we are primarily interested in matters of geometry; in other words: Weight is not of first-line importance. Another fundamental fact can be derived from these considerations: An expandable structure is by its very nature more complicated and less reliable than a rigid structure so that we shall, of course, use it only if we really have to. As we all know, we "have to" in a number of cases and we can expect that this number will increase rather than decrease in the future.

There are two immediately obvious questions with regard to expandable structures: 1. What is their shape, and 2. what material can or should be used.

To talk shortly about the shape, I will show you a few pictures of structures that have been designed in detail at MSFC, such that have been built, and such that are in the early planning phase.

Slides 1 & 2: Solar mirror. This was a design and manufacturing study we did several years ago. The task was to determine limitations and requirements for packaging mirrors within a certain diameter and putting them in operation in space. Considering the diameters which ranged from 20 to 40 feet, we found that only metallic structures could be expected to fulfill the rather severe accuracy requirements for the paraboloidal shape. Shortly after this study was completed, NASA awarded the contract for the sun-flower project to TAPCO, a Division of Thompson-Ramo-Woolridge Inc., which used essentially the same approach for the mirror design by choosing rigid, deployable petals.

Slide 3: The Micrometeroid Measurement Capsule (MMC). The concept was developed in MSFC. Design and manufacturing was contracted to the Fairchild Company. This slide shows the deployment sequence from the fully packaged to the fully deployed condition. Time for deployment 1 minute.

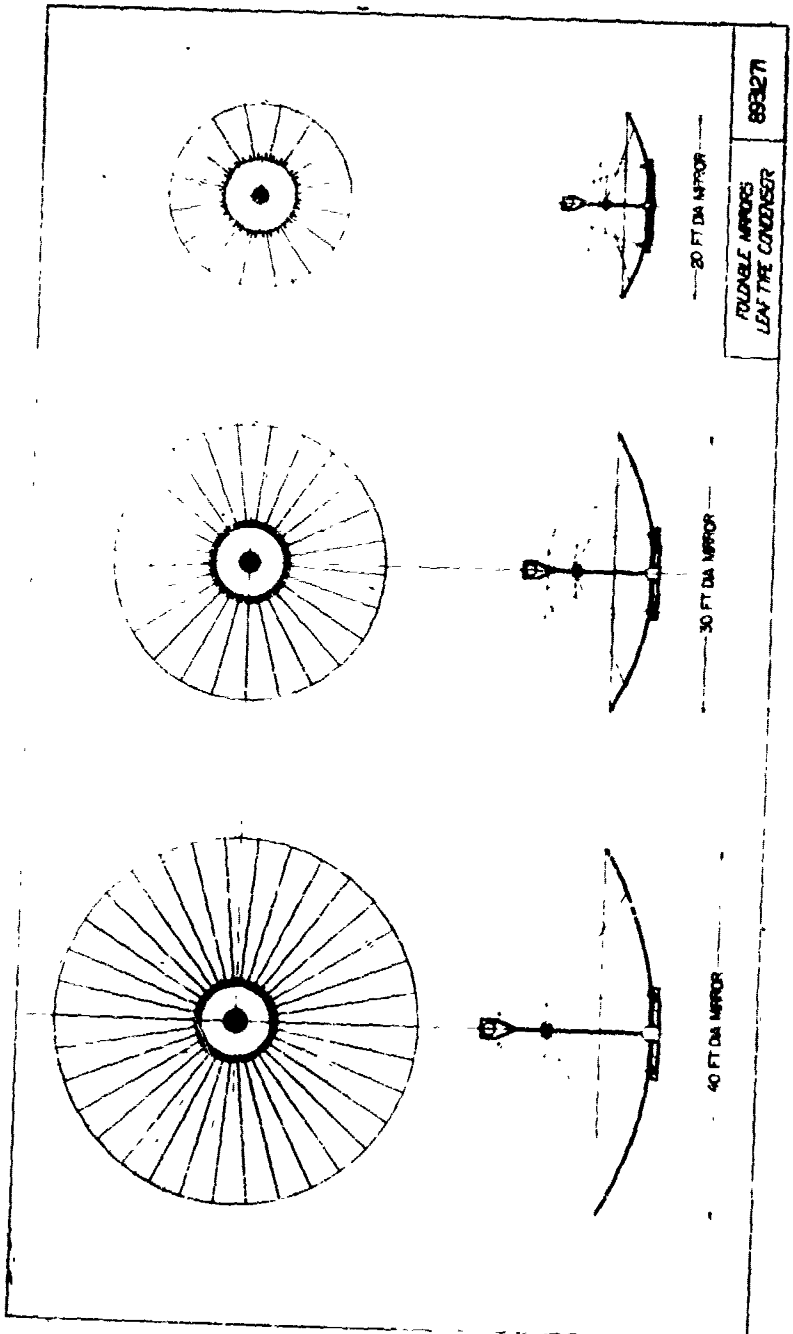
Slide 4: MMC in fully deployed condition. A few data: Total span 96' 4", panel height 13' 8", number of panels 6.5 per wing, actuation by electric motor of 1/60 HP.

Slide 5: To give you a better idea of the actual size of the wings, this slide shows a mockup of the panels.

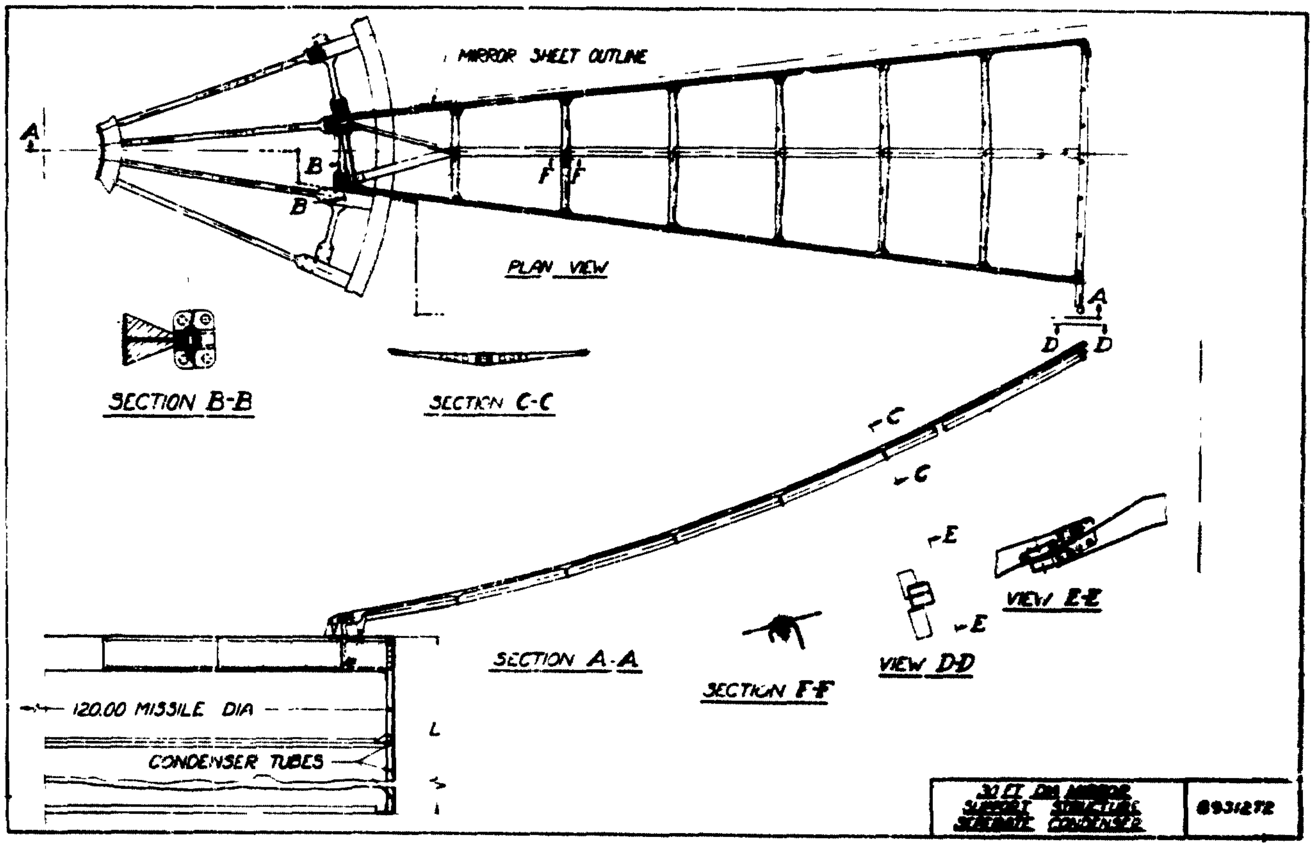
Slide 6: Lunar Landing Vehicle. A very detailed engineering study on all aspects has recently been made by us. The slide is presented in this connection because we have made repeated attempts to find other means than shown here to absorb the energy of the landing impact. The result is that the shock absorbers are built into the main struts of the landing gear. We have made intensive attempts to use such means as large foamed or inflated rings attached to somewhat reduced outriggers, similar rings directly attached to the vehicle; we have even made the assumption that the vehicle could be permitted to overturn and impact on a cushion built around the forward portion of the vehicle. However, all solutions came out heavier and less reliable. It may be mentioned that the requirements for the touchdown were severe: 30° slope plus some velocity component in downhill direction, and c.g. location about 230 inches above the surface.

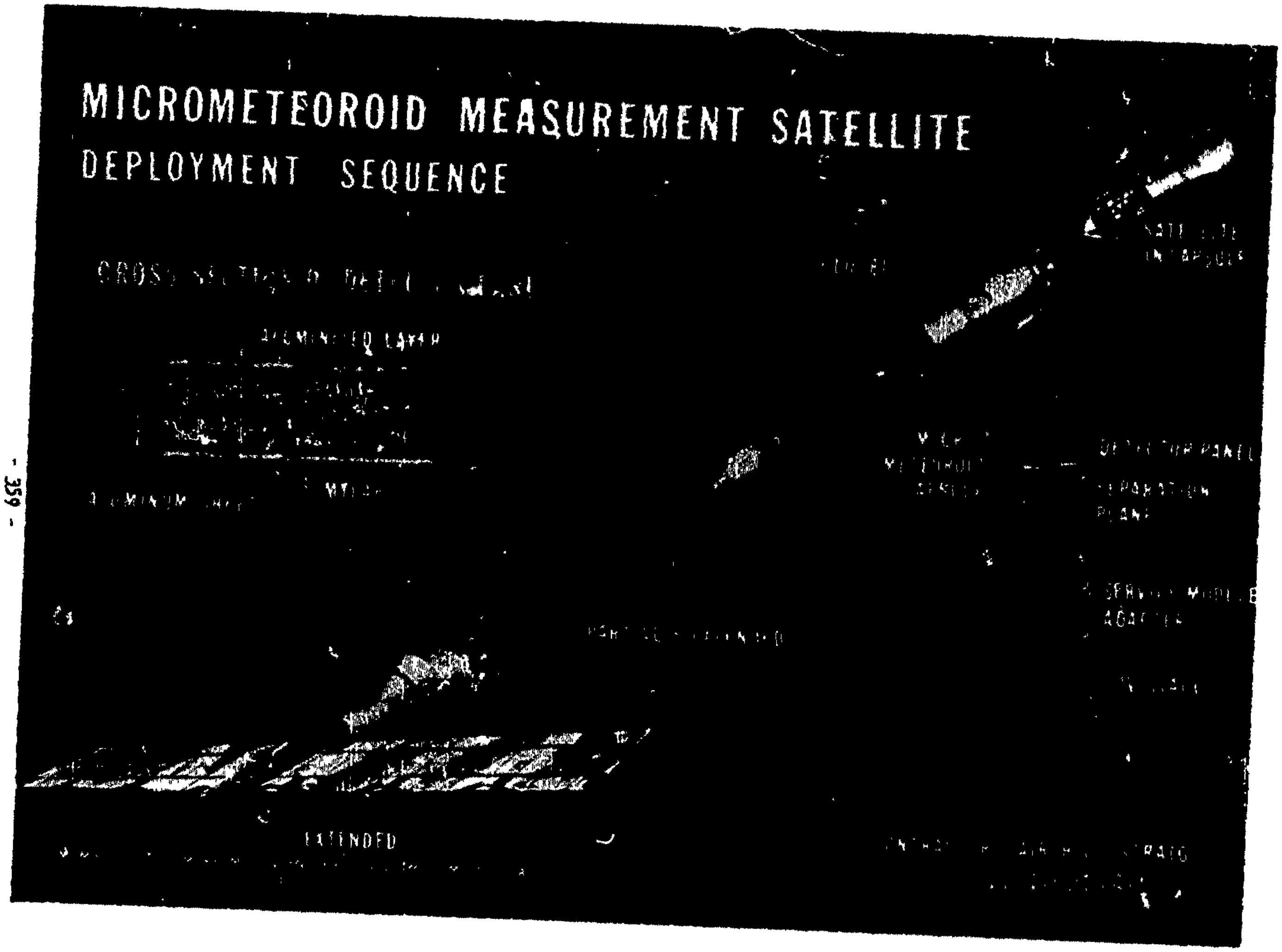
Since MSFC has mission and responsibility for studying the lunar logistics payload effort including lunar bases, we have also begun to study the wide field of lunar shelters.

The term "shelter" covers the whole spectrum from emergency shelters, which may be used for hours or days only, to a lunar base which, to a certain extent, can be considered a permanent installation. The number of people living in such shelters will range from one or two to any reasonable figure.

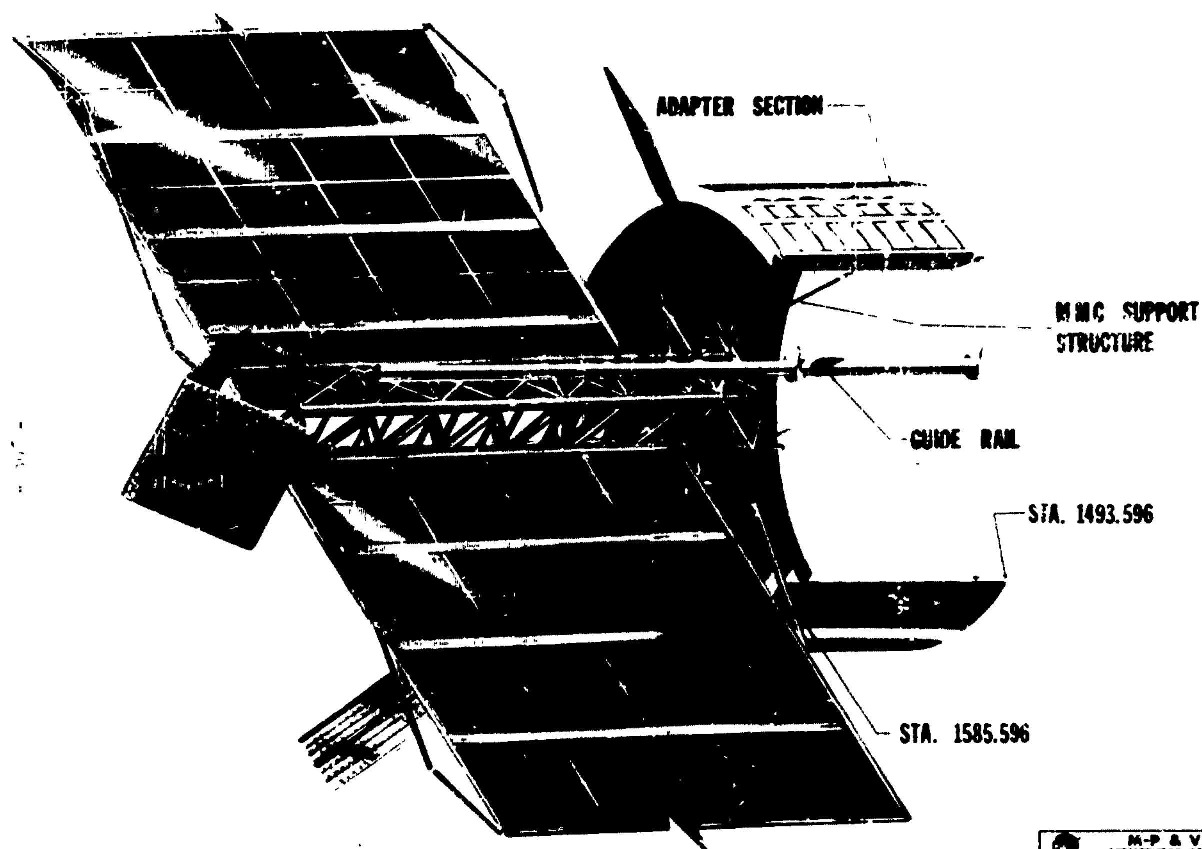


40 FT DIA MIRROR
30 FT DIA MIRROR
20 FT DIA MIRROR
FOLDABLE MIRRORS
LEAF TYPE CONDENSER
893271

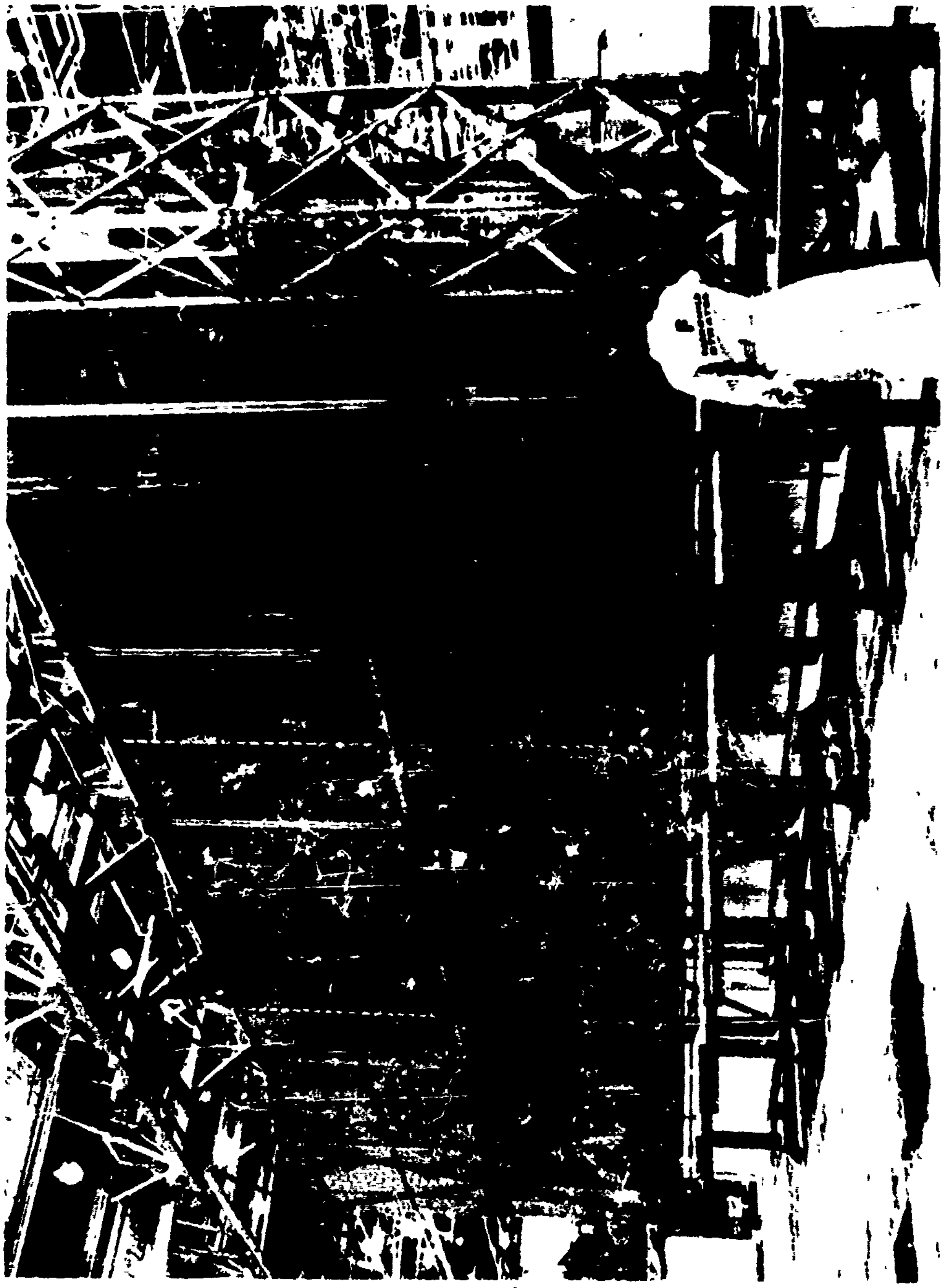

MIRROR SHEET OUTLINE
PLAN VIEW
SECTION B-B
SECTION C-C
SECTION A-A
SECTION F-F
VIEW D-D
VIEW E-E
120.00 MISSILE DIA
CONDENSER TUBES
30 FT DIA MIRROR
SUPPORT STRUCTURE
SCHEMATIC CONDENSER


MICROMETEOROID MEASUREMENT SATELLITE
DEPLOYMENT SEQUENCE
EXTENDED

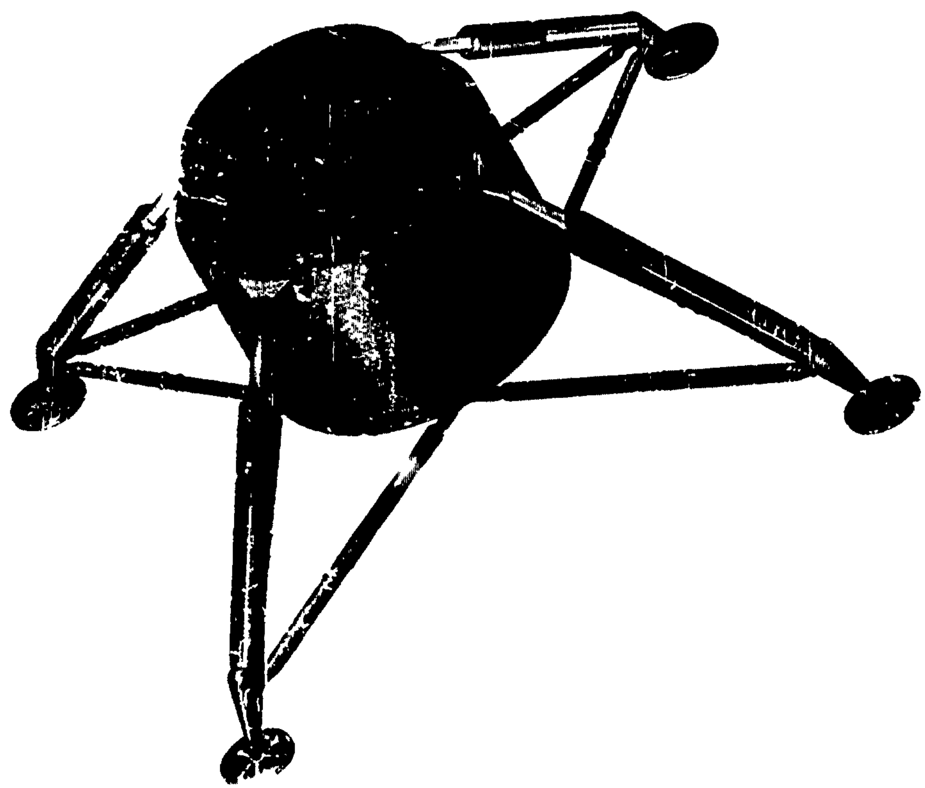
MODIFIED APOLLO BOILERPLATE MICROMETEOROID EXPERIMENT



M-P & VE
STRUCTURES BRANCH



LLV



It is obvious that the size of a shelter, I mean the volume enclosed by such a structure, depends on the purpose, on the mission assigned. We know as a general trend that the amount of space necessary per man should increase with the length of time the shelter will be occupied. We have a very obvious limitation with regard to minimum size: The occupants must be able to stretch to their full length. For an emergency shelter it may be acceptable that this length be provided in horizontal direction, i.e. while the crew is stretched out on their beds. Shelters used for more than emergency purposes must permit standing and some walking.

Aside from such limitations we are free to choose the geometrical shape of the structure but we find that we have only two useful shapes for consideration: Cylinder and sphere. The cylinder can be used in three different ways: as a full cylinder with horizontal or vertical axis, and as a half-cylinder (quonset hut). For obvious reasons the quonset type can be expected to have a very limited application - if any at all. A spherical structure, on the other hand, has a poor volumetric efficiency in comparison to a cylinder having diameter and length equal to the diameter of the sphere. In addition, it appears to have functional disadvantages. We can, therefore, assume that the shelter configuration will be cylindrical with either horizontal or vertical axis. This may also include, of course, a combination of interesting spheres to an elongated, cylinder-like structure.

The following slides show in principle the present thinking with regard to early lunar shelters. All of them are planned for one year life time. They will be occupied intermittently, and will be designed to be transported by the LEM truck.

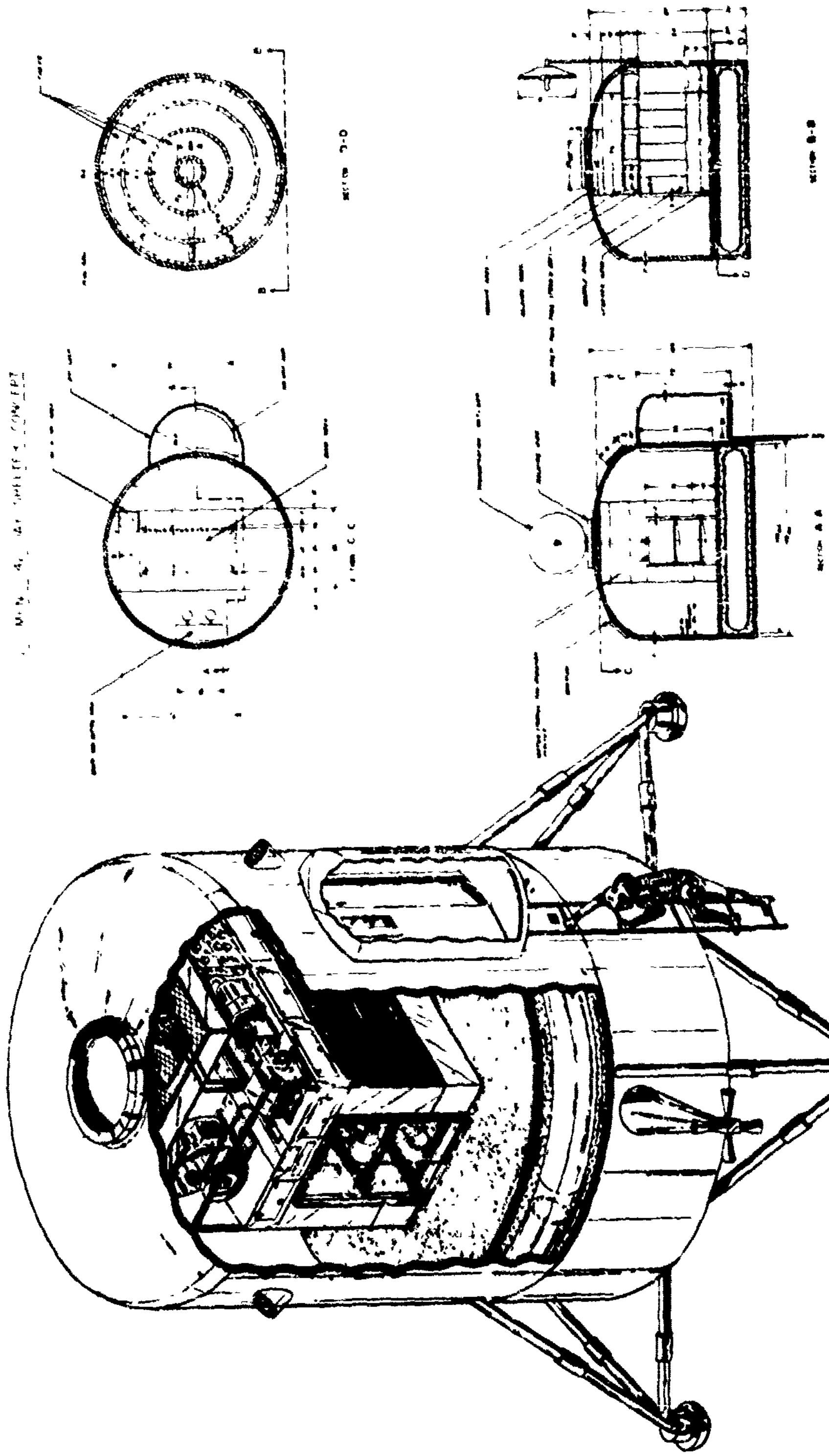
Slide 7: This is perhaps the simplest form of a two-man shelter. The bunks are located at the center of the structure. Instrumentation and equipment are arranged around them. The airlock is shown here on the outside; it can also be arranged to be within the main cylinder of the structure.

Slide 8: This shows a similar shelter design as the previous slide. A lunar roving vehicle is stored underneath the living area.

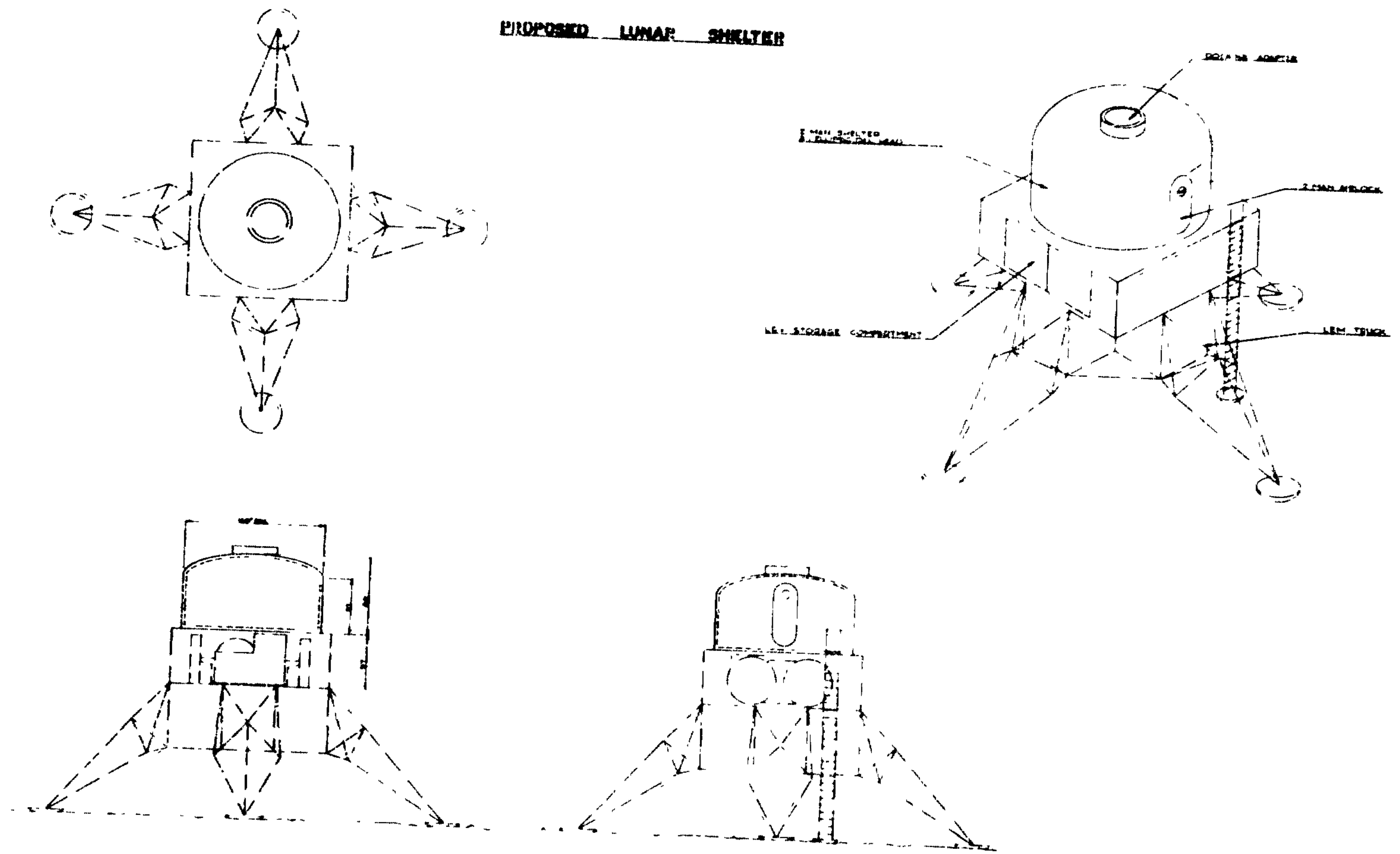
Slide 9: Here we have a cylindrical shelter with a horizontal main axis. Part of the equipment such as fuel tanks and fuel cells are on the outside. Again, a lunar roving vehicle is stored between shelter and LEM truck.

The important characteristic of these shelter configurations is that none of them is an expandable structure. The studies showed that there was no need to use them. I presented these slides only in order to demonstrate that from all we know the early lunar shelters can and will be rigid structures.

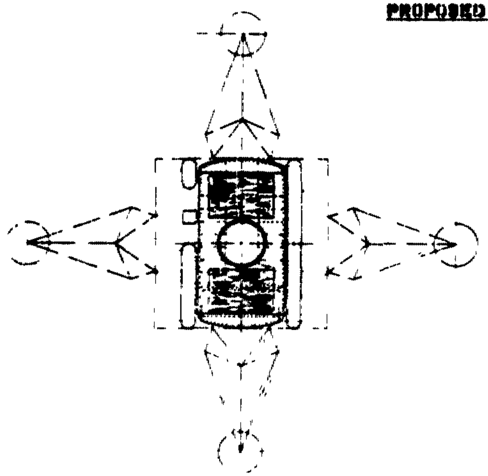
As both the number of people and the stay time on the moon increase, there will be need for larger and consequently different types of shelters.

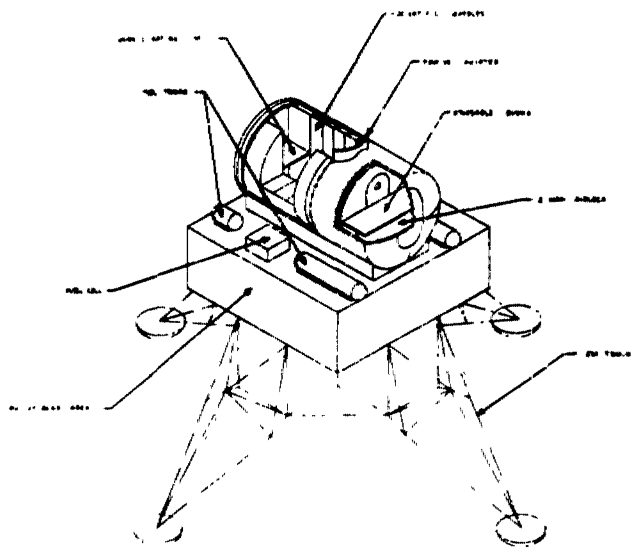


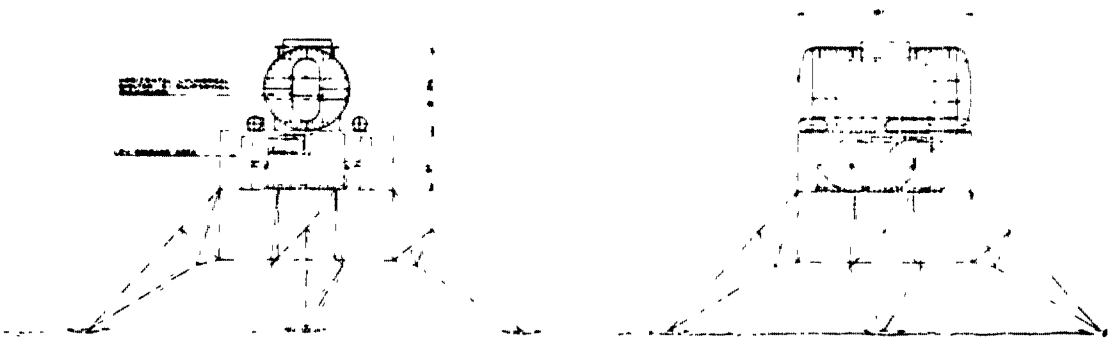
PROPOSED LUNAR SHELTER



PROPOSED LUNAR SHELTER







whether the question of their location - I mean whether they will be located on the surface or in tunnels or trenches - has some influence on considerations of expandability is hard to say at this time; one would probably be inclined to give a negative reply.

All the structures I have shown you have been developed by engineers who were under the relentless pressure of hardware requirements. I would like you to see that, although with these structures and such developed by others, there was made a good start, we still are at the very beginning of developing and using such structures. Ahead of us there is a wide and fascinating field, again a new territory for enormous activities; and we may be inclined to give rein to phantasy.

Now, phantasy is something an engineer certainly needs for any kind of design work. There is no doubt that we need a particularly full measure of it in the fields we are discussing today. But let us understand that besides phantasy - or you prefer to call it imagination, intuition - we need something else equally badly: We need to be realistic. Being engineers we have to dig deeply with both hands into the problems, into the design and manufacturing and functional and environmental problems, if we do not want to fail or to come up with needlessly expensive structures and procedures - expensive both in money and human lives.

We all are aware that it is much easier and much more glamorous to present pictures rather than develop the unfortunately little regarded engineering details. A concept makes sense only to the extent as it recognizes, on one side, the limits of the state-of-the-art (including careful extrapolation) and as it recognizes on the other hand the functional requirements and the influence of the environment.

I suppose that everybody here is fami[illegible] the general [illegible]tion of the environment we have to cope with: [illegible] ture variation, lunar surface, meteorites and [illegible] from impacts on the lunar surface. We ne[illegible] to mention micrometeor[illegible] to demonstrate our lack of quantitative k[illegible] and to show how uncertain the ground still is on which we st[illegible]

As far as the material is concerned [illegible] between metallic structures and non-metallic structures (pl[illegible], fiberglass). The metalli[illegible] structures can be subdivided int[illegible]mpletely rigid and deployable structures (of which only the latte[illegible] interest here), the non-metallic consist essentially of inflat[illegible] [illegible] foamed structures. Of course, both main groups may and do have c[illegible] but still, the main distinction exists - metallic versus non-m[illegible]

As far as application is concerned we know that each group has fields of clear superiority: The Echo balloon i[illegible] quite naturally inflatable and non-metallic, while - equally naturally - lunar landing vehicles are

necessarily metallic. But what should be the answer with regard to space stations or with regard to large lunar shelters and lunar bases which are of growing interest? This answer, I feel, is clearly defined if we consider our pertinent knowledge of materials: We know the metals well enough that we can use them immediately, but we are still in development - and partly in research phases - with the non-metallic materials. Quite obviously, the factor time is important: If we had to design a lunar shelter now or within the next several years, there is no doubt that we would do it in metal. In all fairness we would have to state, however, that beyond this period of the immediate future the answer is much less firm - if we should not leave open the question entirely.

From all I have said so far, you may have the impression that either myself or we at MSFC are against the use of non-metallic structures. This is, of course, not the case. Even if we were so seriously unwise to allow ourselves to have preferences as to materials, we could not afford to overlook or neglect this promising new field. What we have done in each of these cases, however, was an engineering approach using available information, expose it to the full spectrum of manufacturing, functional and environmental requirements, and weigh and compare the results with those of other materials. This procedure will quite naturally be repeated whenever the occasion arises. There is an obvious potential in inflatable and foamed structures but it is apparent that progress has still to be made and experience accumulated before we can with confidence plan on using them. But it should be clear that lack of knowledge should be a stimulant rather than a deterrent for future work.

On the other hand again, if you consider the situation from an engineering point of view, we know that even with metallic structures we shall have our full share of problems with environment and functional requirements - why add material problems if they are avoidable.

To sum up, finally: MSFC's interest in expandable structures is high and real and immediate. As I have shown we are in the process of bringing an expandable structure to actual application and use, we have found with respect to lunar shelters that their early versions can be built in rigid forms; and we are studying what requirements and possibilities the future may bring.

Maintaining the necessary engineering realism we conclude that for the near future metallic structures appear the proper answer. The same engineering realism makes us want, however, to widen and deepen the knowledge in the field of non-metallic materials. Their potential is promising and it is an urgent need to prove and to show how much of it can be converted into the hard facts necessary for the missions that lie ahead of us.

ULTRAVIOLET AND HEAT RIGIDIZATION OF INFLATABLE SPACE STRUCTURES

By

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R. W. Jones
L. B. Keller

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Hughes Aircraft Company

This paper describes the investigation carried out by Hughes Aircraft Company to prove that Mylar film laminates can be rigidized using energy sources available in outer space.

This study showed that both infrared radiation and ultraviolet radiation are practical methods for rigidizing inflatable space structures (such as Echo II).

The objectives of the program were:

1. Small launch volume
2. Minimum weight
3. Good pre-launch shelf life
4. Minimum auxiliary equipment
5. High wall stiffness

Experimental Approach

The approach taken in this program was designed to take maximum advantage of the space environment in the rigidization process. Four environmental factors were considered for this rigidization study; Namely, (1) infrared, (2) ultraviolet, (3) vacuum, (4) gamma radiation. The first two were considered most practical for this program.

The two systems developed were an ultraviolet activated polyester resin and an infrared activated epoxy. Both of these resin systems, combined with a special lightweight fabric reinforcement, produced laminates as many times as stiff as the original Echo I Mylar.

We tried to stay roughly within the weight and size limitations of Echo I but, as such skins could be used for many other inflatable structures, also, the ultimate in low weight and package size was not attempted.

The rigidization process was selected such that there would be no affect on the structure's ability to withstand deterioration between fabrication

and launch. It was decided that, after fabrication, the skin material should show only negligible changes in either initial flexibility or final rigidity in storage for four weeks at room temperature (72 ±2 deg F), two months at 30-40 deg F, two weeks at 80-90 deg F, and three days at 120 deg F. In addition, vibration- and acceleration-resistance requirements were established for launch conditions.

Our analysis of the problem of an optimum structure showed that a laminated design would be most practical, and preliminary test results pointed up the advantage of Mylar film laminates containing catalyzed resins as interlayers. Such laminates can be made to cure to a rigid state when activated by some factor present in the space environment — no ground triggering mechanism is needed. The problem then became one of selecting the best films, catalyst systems, and fabrication methods.

We developed three rigidization systems, all consisting of a laminate of two Mylar skins (one clear and the other aluminized) and a resin-impregnated fabric interlayer, all easily fabricated as thin-skin laminates to a high degree of initial flexibility, and all storable for long periods in the catalyzed condition (see Table 1). For the interlayer, three types of impregnations were used:

1. an ultraviolet-light-activated polyester resin;
2. an ultraviolet-light-activated combination of resin and blowing agent, resulting in a plastic foam;
3. a thermally activated epoxy resin.

Figure 1 illustrates the weights and rigidization achieved by each system.

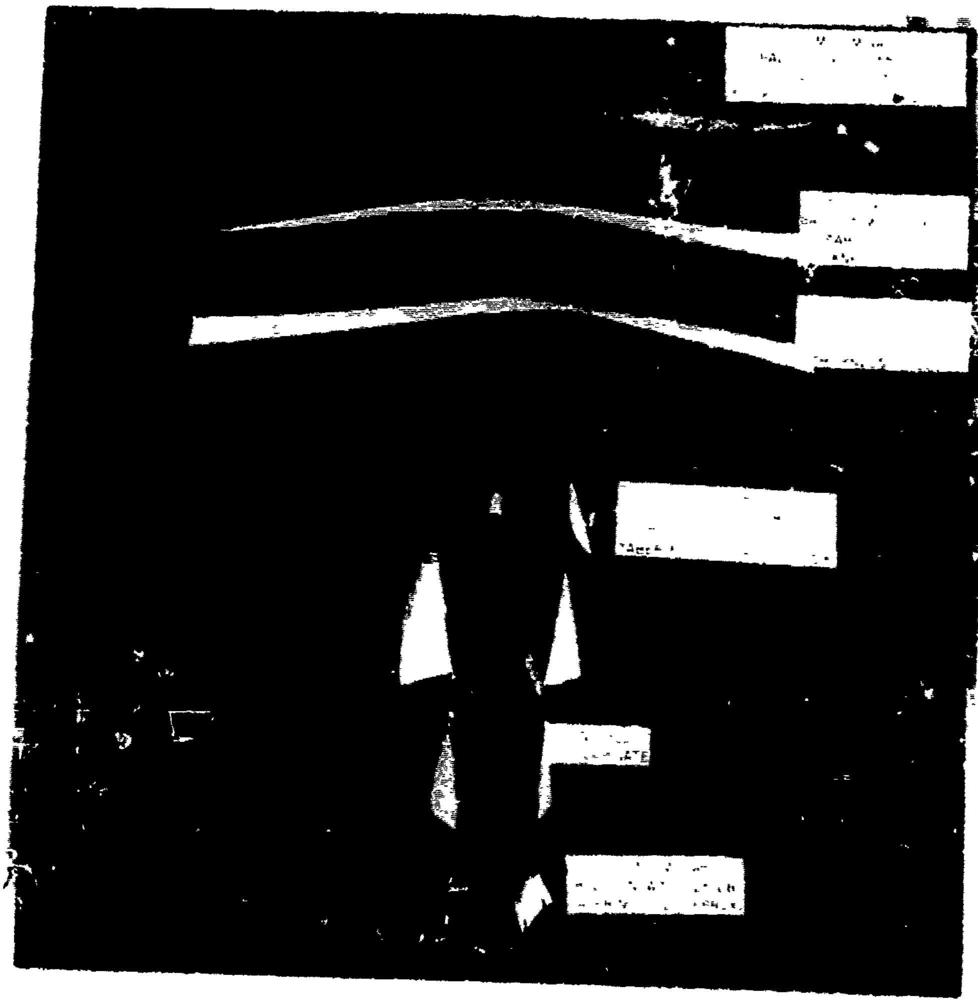
A number of tests were run to determine the stiffness of Mylar-faced laminates containing, at first various amounts of several different resins. We found that a laminate consisting simply of a layer of resin between two Mylar faces was unsatisfactory. For one thing, it was impossible to keep the resin spread evenly; for another, the flexural strength of the layer of cured resin was not much greater than that of an equal thickness of Mylar film.

Since it is known that a fabric-reinforced laminate has considerably more flexural strength than does an equal weight of resin, we next tested nylon hosiery stock as the fabric reinforcement between 1/4-mil Mylar faces. The results for these extremely thin laminates corroborated the theory.

To find the most suitable interlayer fabric, we therefore tested some 44 samples of drapery, hosiery, lingerie, and trimming fabrics — no industrial fabric available was light enough. A special Dacron marquisette was selected which was 0.0044 in. thick and weighed 0.45 lb per 1000 sq ft.* This was neither the lightest nor the thinnest fabric but it offered the best combination of properties in terms of handling strength, final rigidity, weight and thickness.

*Marquisette, n. A sheer, somewhat lustrous cotton fabric; also a sheer light silk fabric for dresses -- Webster's New Collegiate Dictonary.

Figure 1. Illustrating the relative rigidities of the various systems.



	Weight per 1000 Sq Ft (lb)	Taber Stiffness No.*
1/2-mil Mylar (Echo I)	3.6	0-1/2
Polyester laminate	12.27	30-50
Epoxy laminate	12.27	40-60
Polyester foam laminate	17.5	75-150

*Empirically equated to flexural rigidity (in inch-pounds) divided by 0.396×10^{-4}.

Table 1. Weight and stiffness of inflatable materials.

Tests also were carried out to determine the resin or resins to be used in making the laminate. At least one of our systems was to be actuated by ultraviolet radiation. Polyester resins, all of which can undergo ultraviolet catalysis, were an obvious choice. A series of tests were made of the reactivity of various combinations of ultraviolet catalysts with polyester resins modified by styrene, diallyl phthalate, and triallylcyanurate. The styrene modified resins gave the highest flexural strengths. Unfortunately, because of the rapid vaporization of the styrene monomer, this material is not suitable for either long, unsealed storage or use at high altitudes.

Prolonged storage tests on the ground plus short exposures at simulated extreme altitudes showed that triallyl-cyanurate-monomer-modified resins would be satisfactory. To get maximum strength from these resins, we developed a special ultraviolet- and heat-actuated catalyst system.

The flexible laminates were made by impregnating the light fabric with a solution of the catalyzed resin. Upon loss of solvent, the resin then firmly adhered to the 1/4-mil Mylar faces. These laminates met all the flexibility, storage, and vibration- and acceleration-resistance requirements for catalyzed, uncured materials. Tests with an ultraviolet source showed that they would reach about 80 percent of their full strength within 20 minutes after the start of exposure. The full strength was attained in 24 hours.

To gain the ultimate in flexural strength, we fabricated foam-filled sandwich structures, using the same polyester resin system as before but with a diazo blowing agent added to it. Under ultraviolet radiation, the blowing agent decomposes and releases nitrogen gas, causing the resin to foam and, at the same time, to polymerize. The resulting structures were about four times as stiff as the original laminate; however, they were also about 25 percent heavier.

Our third laminate, the epoxy resin system, equals the uncured, solid polyester system in flexibility, storability, and all other properties, except that it has a somewhat better stiffness-weight ratio after curing. Unfortunately, about 250 deg F is needed to start the epoxy curing reaction. This

means that the infrared absorptivity and emissivity characteristics of the laminate skins must be adjusted to produce this temperature. Because two different skins could be used (clear and aluminized Mylar), tests at UCLA's Thermal Radiation Lab showed it was possible to arrange the configuration so that the activation temperature could be reached in either system.

Our final choices were the two configurations using the polyester resin and the epoxy resin (Figures 2 and 3). Figures 4, 5, and 6 show the change in flexural rigidity and a tensile strength with temperature for these systems after cure under conditions considered typical for an inflated structure like the Echo I. The increase in rigidity after an initial drop-off can be attributed to post-curing of the resin systems. The loss of weight and strength as a result of high-vacuum effects proved negligible in a two-week test period. The tear strength (as measured by the ASTM D1004 method), showed an increase of about 500 percent over that of the Echo I's 1/2-mil Mylar for all laminated structures. These results substantially agree with the data on similar materials in the literature.

This kind of rigidization, we believe, can also be applied to thicker materials, which may then be used in structural members.

The use of ultraviolet as a rigidizing mechanism has a number of advantages. They are.

1. An indefinite shelf life is available.
2. No activation mechanism is required.
3. There is no excess weight requirement such as in the case with a solvent boil-off system or a gelatin system.
4. No gaseous catalysts are required.
5. The resins are commercially available and inexpensive.

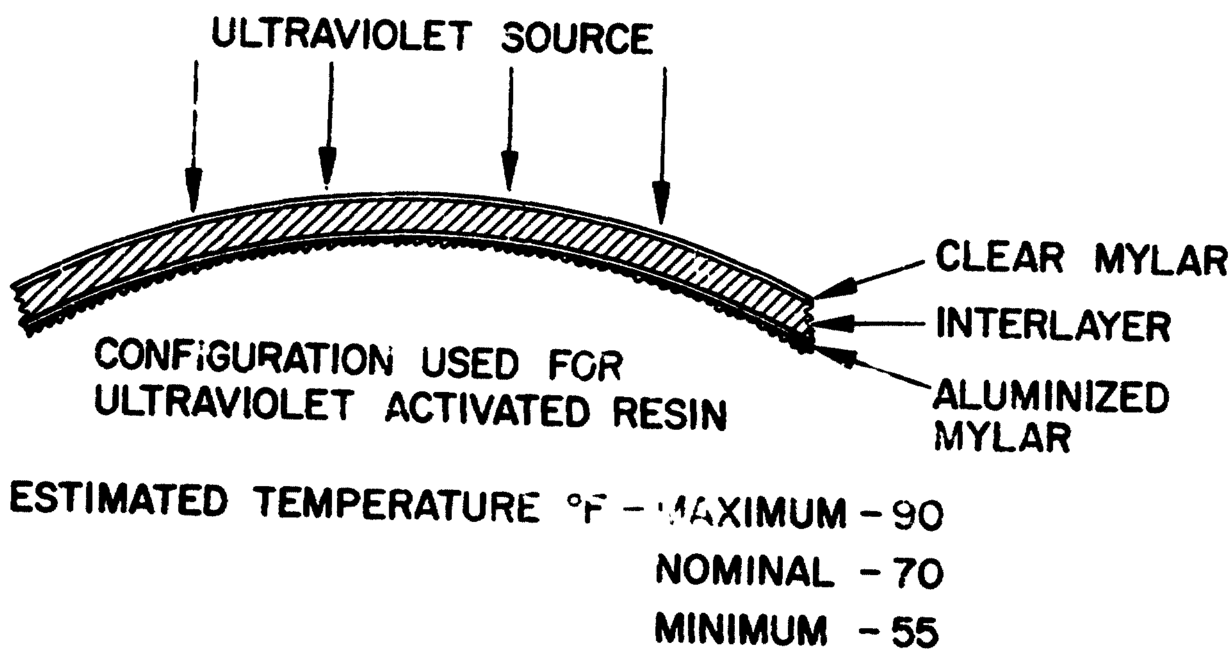
The two possible drawbacks to U-V cured systems are:

1. U-V source must be available, i.e., sun orientation is required.
2. Long time U-V exposure may cause further degradation of the polymer film.

With respect to the second point, it has been pointed out by Dr. Pezdirtz of NASA Langley and others, that in many cases the space structure being rigidized may be coated with aluminum on one side (such as a solar concentrator). Thus the aluminized surface will serve to block further U-V degradation.

Another proposed technique to increase the U-V resistance would be the incorporation into the system of a material having a variable U-V absorptivity. This material would have a low initial U-V absorptivity which allows U-V catalyzation, and after a short period changes to a high U-V absorptivity acting as a protective agent.

Figure 2. Configuration used for ultraviolet violet activated resin.



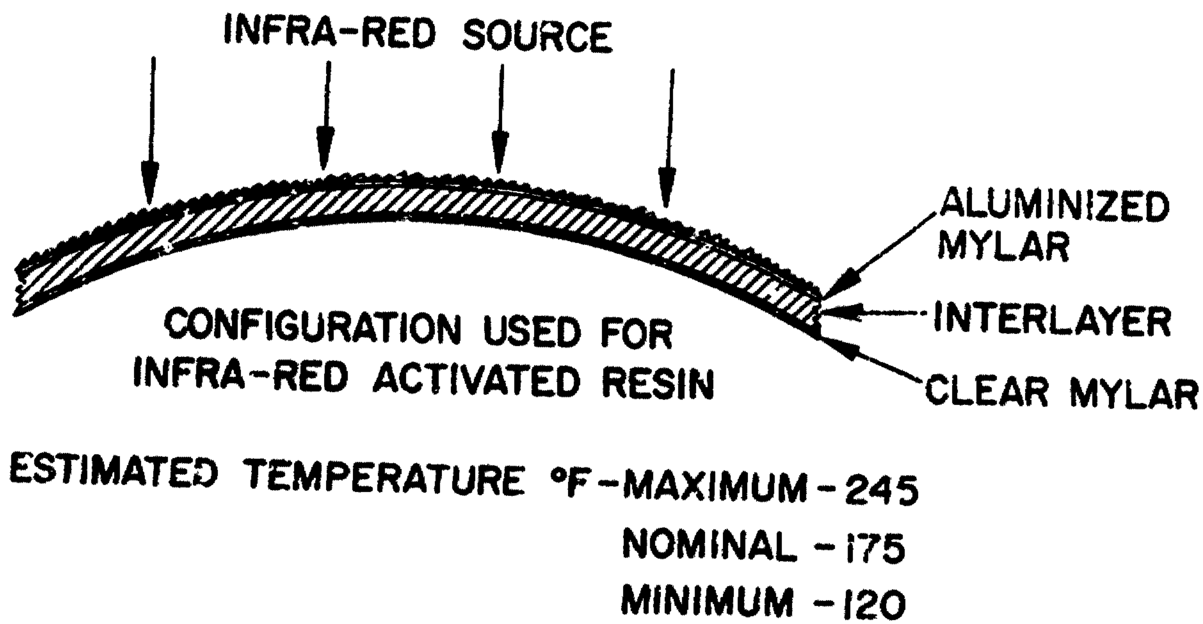
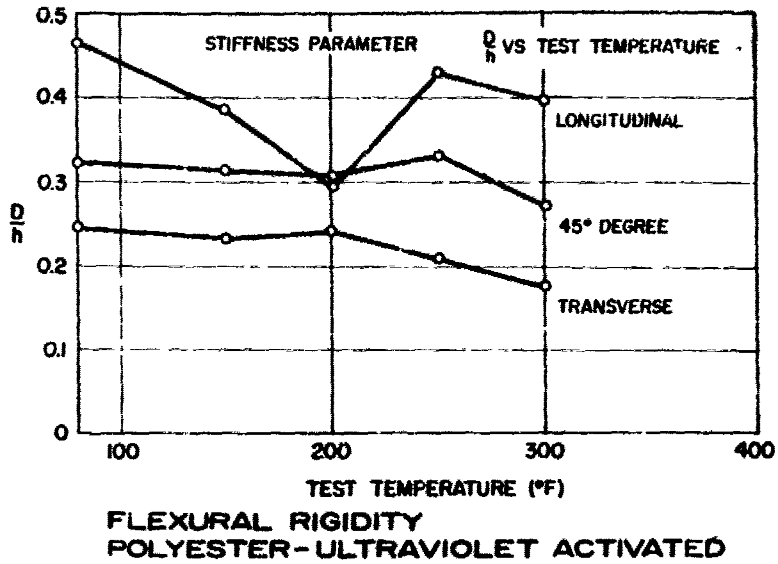


Figure 3. Configuration used for infrared activated resin.

Figure 4. Flexural rigidity of U.V. activated polyester laminates at various temperatures.



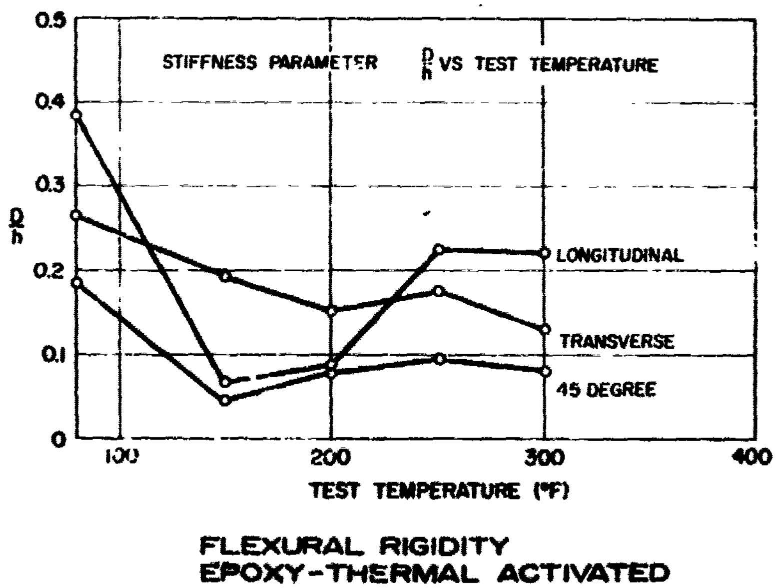
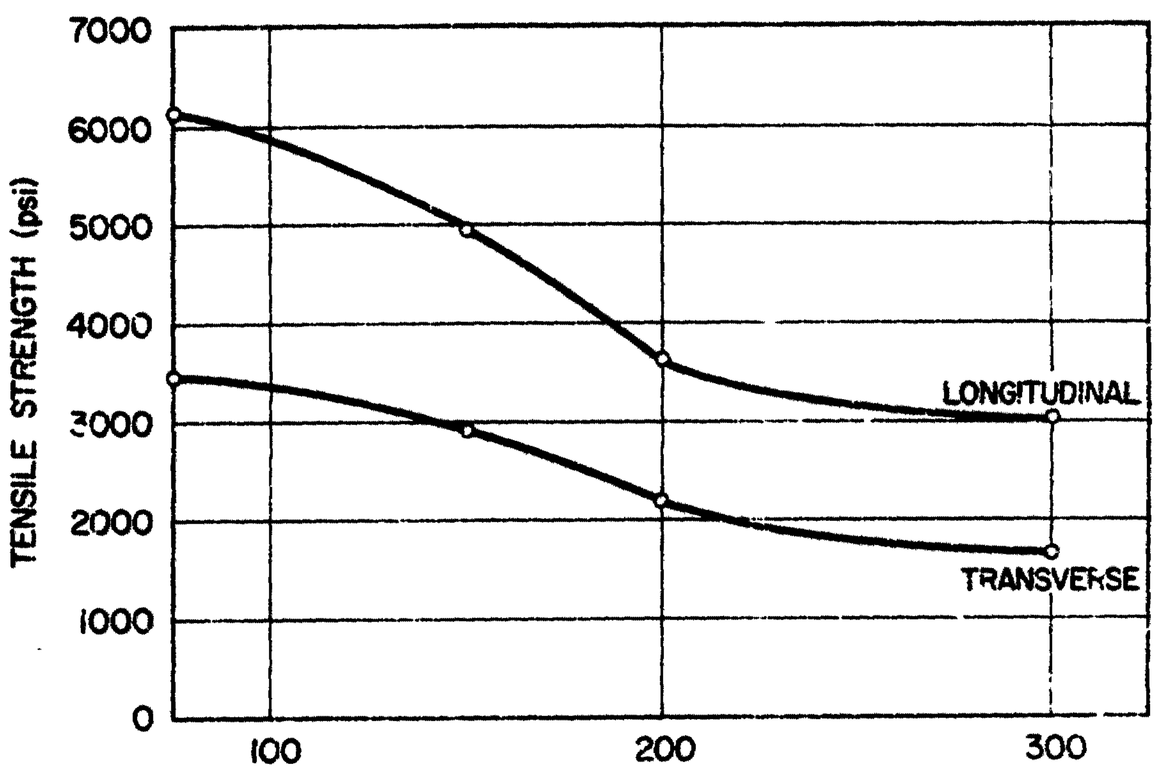


Figure 5. Flexural rigidity of thermally activated epoxy laminates at various temperatures.

Figure 6. Variation of tensile strength with temperature for TAC polyester laminate.



TENSILE STRENGTH POLYESTER-ULTRAVIOLET

Such a material has been described, and is commercially available. The compound resorcinol monobenzoate does not absorb ultraviolet radiation until it has been exposed for some time to U-V.

Thus, a fresh 0.005 percent solution of resorcinol monobenzoate in isopropyl alcohol transmits 50 percent or more light above 290 mμ, but after 16 hours this wave length is increased to 360 mμ. The ultraviolet light may be causing rearrangement of the resorcinol monobenzoate to 2, 4-dihydroxybenzophenone, a good inhibitor against UV damage. Theoretically, the resorcinol monobenzoate would have to be separated from the resin, since phenolic materials tend to inhibit polymerization of the polyester resin. However, in a test of a U-V catalyzed resin containing 1 percent of resorcinol monobenzoate, there appeared to be no cure inhibition at all. At the time of writing tests are being conducted on such materials to determine the effect on U-V degradation.

The principal disadvantage to the use of an infrared activated system (which could be made very ultraviolet resistant) is the fact that for activation a fairly high surface temperature is required. This then presupposes a surface which has a relatively high absorptivity and a low emmissivity. Unfortunately, such a structure, after rigidization is complete, would remain at the high temperature unless a coating was developed with a variable $\frac{\alpha}{\epsilon}$. Such a coating would be used to initially result in a high surface temperature and later would change to a low surface temperature. This would result in increased strength in the structure as well as cooler internal parts, better liveability, etc. Such a development would then make practicable the utilization of a thermally activated system such as the epoxy or a heat activated polyester resin or polyurethane foam. Hughes Aircraft Company is currently investigating variable coatings.

A third procedure, in which only moderate temperatures are required, would be to use an initiator which decomposes at a relatively low temperature. This substance would initiate the decomposition of a free radical source such as benzoyl peroxide which would cause polymerization of a polyester resin. Such an initiator may be a compound like azobisisobutyronitrile which decomposes at 140 to 190°F. With a surface having an $\frac{\alpha}{\epsilon}$ ratio which would assure the attainment of these temperatures, the structure could be allowed to remain at that temperature without loss in strength. By this system, good ultraviolet protection could be obtained from the coating designed to reach the required temperatures. Tests would be required to determine what storage life is available from such a system.

The successful development of either or both of these systems to resist successfully space environment deterioration could then lead to a considerable increase in the utility of expandable structures.

It is hoped that other investigators will explore the further application of heat and ultraviolet rigidized space structures.

NOTE:

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LIST OF ILLUSTRATIONS AND TABLES

THE POTENTIAL OF EXPANDED ALUMINUM

MESH FOR INFLATABLE RIGIDIZED RADAR

REFLECTIVE DEVICES

L. J. Boler

Viron Division

Geophysics Corporation of America

I. PHYSICAL DESCRIPTION OF EXPANDED MESH

A. Expanded Mesh Fabrication

Expanded mesh is fabricated from sheets or coils of material with no waste.

A typical example of the magnitude of expansion would be as follows:

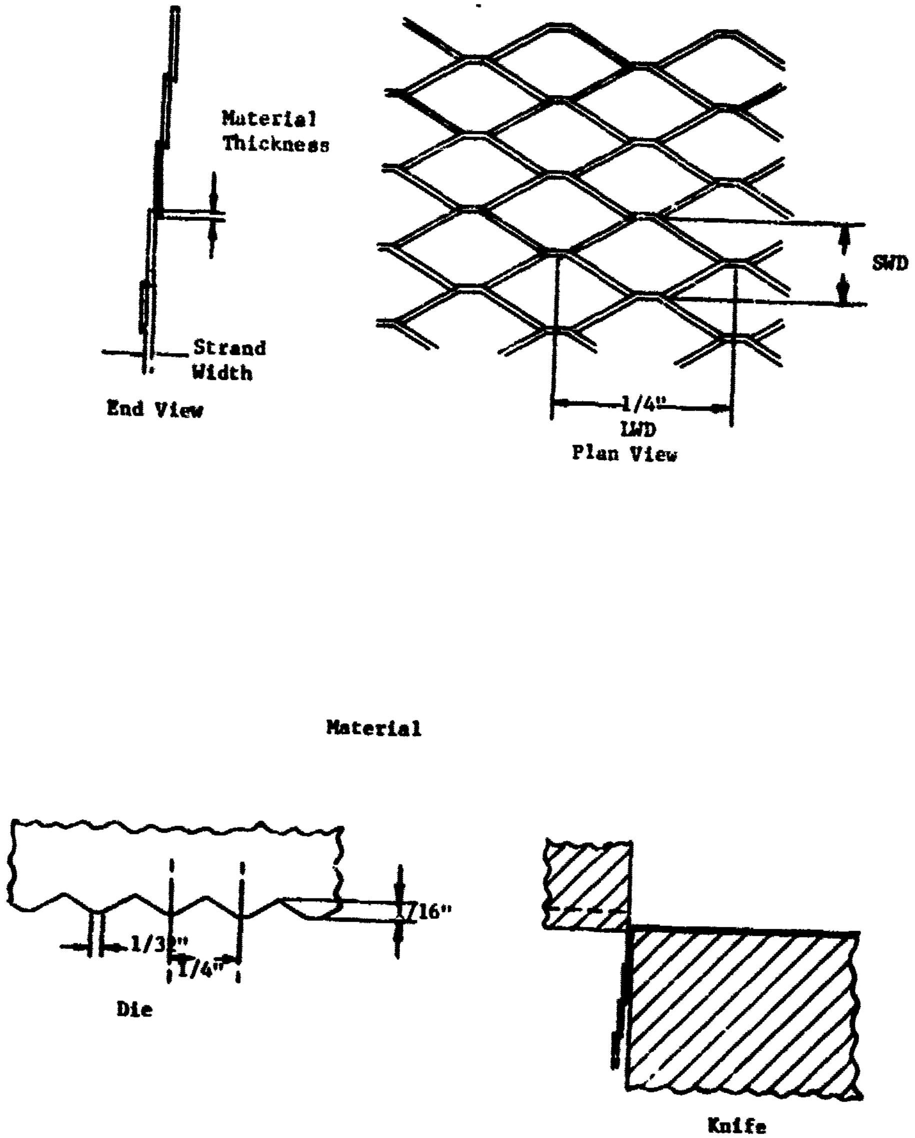
Parent Material: .003" thick aluminum foil one foot square.

Expanded Material: an expanded product with 94% open area with a strand width of 0.010" would be one foot wide could be 6 to 12 feet long.

Many materials can be expanded. Some examples are shown in Table I.

Metals:	Aluminum
	Lead
	Brass
	Copper
	Silver
	Platinum
	Nickel
	Steel
Non-Metals:	Polyethylene
	Teflon
	Any plastic that is not brittle
Laminates of Metals and non-metals	

TABLE I. MATERIALS THAT CAN BE EXPANDED



Typical Die and Knife

FIGURE 1

The expanding process utilizes the entire sheet of material that reaches the expanding dies, that is, there is no waste, no blanks or punched out parts.

The expanded mesh is simultaneously slit and stretched by the shaped dies. The shaped dies determine the dimensions of the openings in the mesh.

An understanding of the process can be obtained by studying Figure I.

Special desired characteristics of the finished material can be obtained by varying the SWD, LWD, strand width and feed material thickness.

LWD - Long way of the diamond - measured from the center of one joint to the center of the next joint. This dimension is determined by the dies.

SWD - Short way of the diamond - measured from center to center of the joints in the machine direction. This dimension can be changed slightly by adjusting the vertical travel of the dies. The maximum SWD is twice the die depth minus twice the material thickness.

Strand Width - The strand width is the width of the material that is slit from the original material with each pass of the dies. The strand width is usually equal to or greater than the original material thickness.

Original Material Thickness - The original metal thickness becomes the strand thickness. Thus the weight of a particular mesh configuration is directly proportional to the original material thickness. The general range in micro mesh for this dimension is from .001" to .010".

Some of the possibilities are illustrated in Table I.

TABLE II
HOW TO OBTAIN SPECIAL CHARACTERISTICS IN EXPANDED MESH

	Change the following item as indicated			
Desired Characteristics	SWD	LWD	Strand Thickness	Material Thickness
Strength	Decrease	Decrease	Increase	Increase
Reduce Weight	Increase	Increase	Decrease	Decrease
Increase Rigidity	(slight effect) Decrease	(slight effect) Decrease	(large effect) Increase	(large effect) Increase
Radar Reflectivity	Adjust	Adjust	Increase	Increase
Low Drag	Increase	Increase	Negligible	Large-effect

F. Possible Finishing Operations

Different applications will require different configurations, the basic adjustments that can be made have been discussed in Section 1A, however, other finishing operations are available.

1. Rolling

The mesh can be rolled flat as shown in Figure 2

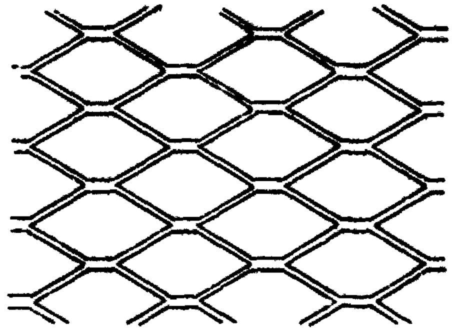


FIGURE 2
EXPANDED MESH ROLLED FLAT

2. Coating

A coating can be applied easily by a dipping process. Coatings could be applied to enhance appearance, to obtain thermal properties, or to obtain corrosion resistance.

C. Vehicle Construction

Most applications of the expanded metal either as radar reflector or in other applications have and will require seams so that spheres, cylinders, etc. of the proper size can be fabricated from flat pieces of the mesh material.

The most readily apparent method of erecting these structures is by inflation.

Inflation requires a membrane to hold the inflation gas or vapor from escaping during the inflating process. Polypropylene and Mylar plastic films would be logical choices for the inflation membrane material. Some of the factors affecting Polypropylene and Mylar films have been investigated by Viron under Contract NAS 5-1888 of the NASA. Some of the results of these investigations are discussed below:

1. Physical Characteristics of Mesh and Laminate

The weight and dimensions of the meshes and their laminates being used was experimentally determined and are shown in Table III. A template was used to give consistent accurate sample sizes. Weights were established to within one per cent. The weight and dimensions of mesh and laminate will vary some from batch to batch and within a batch because of manufacturing tolerances on the mesn and the film, the variation of the amount of adhesive used in the lamination, and the variation of tensions during the laminating process.

a. Thermal Shock and Flexure Test

Two sets of samples were prepared for the thermal shock test. Each set consisted of six samples of each

TABLE III
PHYSICAL CHARACTERISTICS OF MESH AND LAMINATE

Mesh	Film	Aluminum Alloy	Aluminum Thickness (mils)	Strand Width (mils)	Strand Length (inches)	Small Angle of Diamond (degrees)	Unit Weight (#/sq.ft)	Approx. Packing Density (cu in/#)	Estimated Sphere Weight (Lbs.) 135' Dia.
Echo I	(0.5 mil metallized Mylar)						0.00376	70	(233)
A12	(Laminate of 0.18 mil aluminum on each side of 0.35 mil Mylar)						0.00786		473
#1	---	5052	3	6	.226	61.0	0.00224	---	135
#1	---	1145	3	6	.225	56.7	0.002[illegible]4	---	141
#1	0.2 mil Polypropylene	5052	3	6	.222	66.2	0.00589		357
#1	0.25 mil Mylar	5052	3	6	.227	59.2	0.00720	150	435
#1	0.2 mil Polypropylene	1145	3	6	.22[illegible]	53.8	0.00495		302
#1	0.25 mil Mylar	1145	3	6	.226	[illegible].2	0.00692		418
#2	---	5052	2	5	.153	45.0	0.00246	---	148
#2	---	1145	3	4	.146	[illegible]	0.00271	---	163
#2	0.2 mil Polypropylene	5052	2	5	.150	62.4	0.00591		359
#2	0.25 mil Mylar	5052	2	5	.146	73.4	0.00732		442
#2	0.2 mil Polypropylene	1145	3	4	.150	53.8	0.00532		324
#2	0.25 mil Mylar	1145	3	4	.146	55.2	0.00683		413

NOTE: #1 Mesh is .405" LWD by .215" SWD
#2 Mesh is .280" LWD by .112" SWD

mesh laminate, two samples with single-taped seams, and two samples with double-taped seams. The samples were placed between wire screen and successively submerged in liquid nitrogen (-195.5°C) and boiling water (+100°C). The test was continued for a total of thirteen cycles, each cycle consisting of a submersion of 5 minutes in nitrogen and 5 minutes in boiling water. After completing the thirteen cycles the samples were allowed to air dry.

The samples were inspected under a microscope. No breaking, cracking, or delamination was observed. Some discoloration and some wrinkling was evident on all of the samples. The taped seam did not crack, break, or discolor. Its mechanical strength was not degraded.

Two-inch wide flexure test strips were clamped around an 8" aluminum ring. Samples were submerged 25 times for ten seconds alternately in water and liquid nitrogen. No cracking, breaking, or delamination of samples occurred. Slight discoloration and wrinkling was observed.

b. Seam Construction

1) Method of constructing seams for the mesh laminate material are currently under investigation. The normal seam construction used with the plastic portion of the laminate will probably yield sufficient strength.

D. Sphere Construction

Utilizing a taped seam construction, eight one meter spheres and five 12½ foot spheres have been constructed and structurally tested. Some of the test results are summarized in Table IV.

TABLE IV
AVERAGE STRAIN IN PERCENT OF MYLAR MESH LAMINATE SPHERES

Strain Location	Sphere Skin Stress		
	4000 PSI	8000 PSI	Ultimate
Longitudinal			
Equator	0.91	1.63	3.45
45° up from Equator	0.87	1.55	2.97
On Gore Adjacent to Polar Cap	0.73	1.15	2.20
Transverse			
Equator	0.57	1.20	3.51
45° up from Equator	0.61	1.24	2.90
On Gore Adjacent to Polar Cap	0.82	1.23	2.47
Circumferential			
Equator	0.41	0.81	2.17
Polar	0.46	0.81	1.68

This work illustrates that a mesh sphere vehicle can be manufactured using either Mylar or Polypropylene plastics.

E. Packing Density

Measurements of displaced volume have been made on folded 12½ foot spheres fabricated with #2-1145 - 0.15 mil mylar mesh laminate. The folded spheres were placed in polyethylene bags, evacuated, and sealed to approximate a packing pressure of one atmosphere. The experiment resulted in a packing density of .00369 cubic inches of displaced volume for each square inch of surface area. Extrapolation of this figure to a 135 foot sphere gives a volume of 17.4 $ft.^3$ for a mylar mesh laminate.

Both mylar mesh laminate and polypropylene mesh laminate have also been stacked and subjected to pressure. Plots of thickness per layer with pressure are shown in Figure 3.

Polypropylene mesh laminate occupies from 55 to 70% of the volume of mylar mesh laminate. A conservative volume estimate for a 135 foot polypropylene sphere based on this information is 12.25 $ft.^3$. One would expect that 135 foot sphere can be folded and packed in a volume from 85% to 95% of the extrapolated volume based on measurements made on a 12½ ft. sphere. Applying this factor results in a balloon volume estimate for a 135 foot polypropylene mesh laminate sphere of 11.0 cubic feet.

Calculations of canister radius versus factored balloon volume for various balloon stack heights have been made. This family of canisters is included in figure 4.

An estimate of the necessary canister radius for a 135 foot polyester sphere, assuming a balloon stack height of 10 inches is slightly less than 25 inches.

It is thus adequately demonstrated that the expanded aluminum mesh can be used to manufacture spheres which can be used as radar reflectors.

II. EXPANDED MESH AS A RADAR REFLECTOR

A. ASD Testing

Considerable testing has been accomplished by ASD to determine the efficiency of Metal Grid Spheres as Reflecting Bodies. ASD technical note 61-85 dated November 1961 by YLO E. Stahler reports that "As Passive Reflectors," Grid Spheres can replace full metallic surface spheres as long as the mesh dimensions are kept smaller than 1/8 of the communication wave length.

The testing reported on in the above study was conducted with an 18 inch Painted Grid test sphere or a 22-3/4 inch sphere covered by a grid of thin wires, primarily, .010 inch in diameter.

Viron has conducted limited tests 12½ ft. diameter mesh spheres under contract NAS 5-1888. The results were inconclusive and this is an area where additional efforts definitely required.

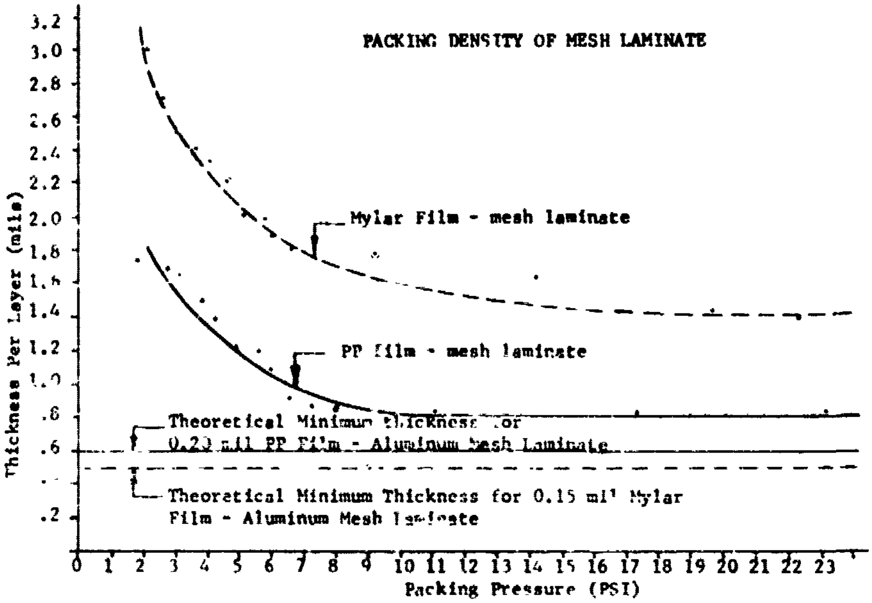
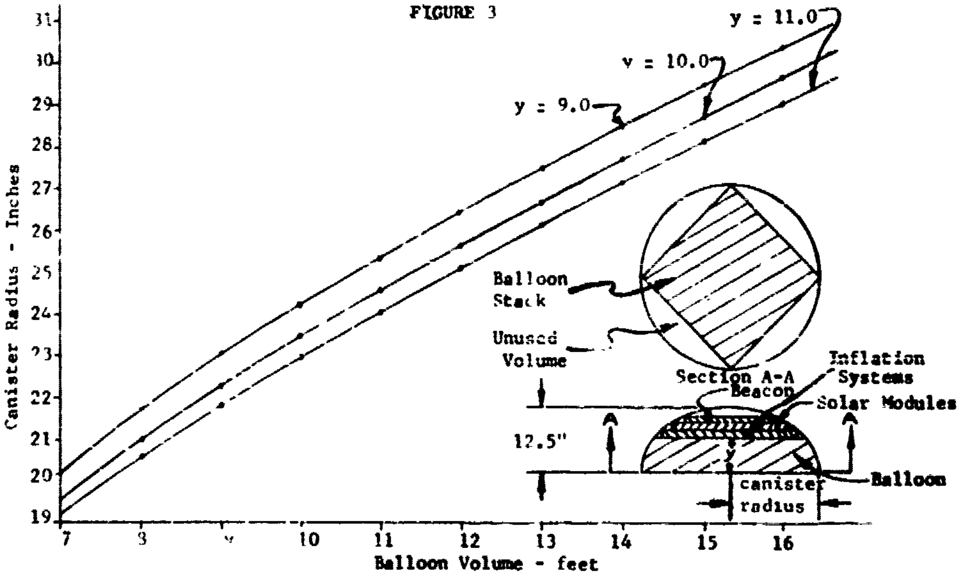


FIGURE 3



CANISTER RADIUS VS. BALLOON FOR 135 FOOT SPHERE

8

FIGURE 4

III. EXPANDED MESH RIGIDITY

It is essential that the mesh vehicle attain a designed shape and then retain that shape for the flight design period. This end result can be accomplished in the case of the mesh laminate construction because of the fact that any metal that would be of interest as a radar reflector will have a yield pt. that can be exceeded by some method of internally pressurizing the mesh structure.

Once this yield point has been exceeded the geometric shape is essentially fixed as long as the vehicle is operating in the environment of space and has been properly designed to withstand the forces of solar pressure, atmospheric drag, micrometeorites impacts.

The following discussion is directed toward comparing the stiffness of the expanded aluminum mesh laminate with other materials such as the metallized mylar used in Echo I and the aluminum foil, mylar, aluminum foil laminate used in Echo II.

A simple cantilever beam of length l end loaded with a load W has a maximum deflection of $\frac{Wl^3}{3\,EI}$ where E is the modulus of elasticity of the material and I is the amount of inertia of the cross section with respect to its neutral axis. The EI product occurs in the denominator for the deflection expression of many other beams with more complex loadings and supports. Since deflection is the inverse of stiffness, and deflection is inversely proportional to the EI product, the EI product of any construction can be used as a criteria of stiffness.

The EI product of the mesh laminate and Echo I and A-12 materials was first determined analytically. It was also determined experimentally and compared with its analytical value.

The mathematical determination of the EI product of mesh material becomes complex since it is not homogeneous or isotropic. Dimensions and physical properties of mesh laminate change from point to point.

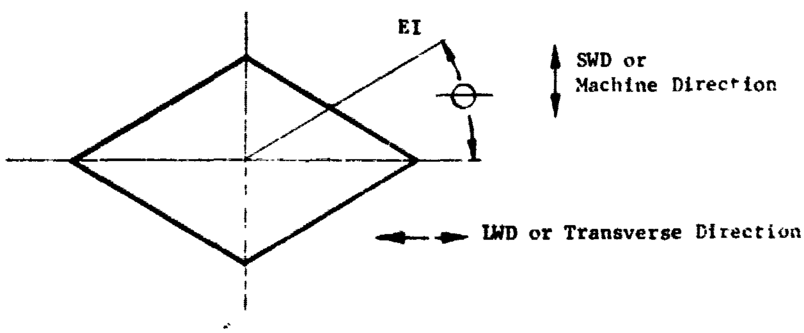


FIGURE 5
GENERAL MESH

For the general mesh we can say that

$$EI = f\ (\theta, X_1, X_2, \ldots X_n)$$

Where θ is the direction along which the EI product is determined, and X_n is the nth prameter of the mesh which affects stiffness.

Only two cases, $\theta = 0$ (or 180°) and $\theta = 90°$ (or 270°) are considered in the analysis. The stiffness in one direction is the same as that 180° from it. The direction when $\theta = 0$ (or 180°) is referred to as the transverse direction LWD (Long Way of the Diamond); the machine direction SWD (Short Way of the Diamond) is the direction $\theta = 90°$ (or 270°).

The analysis performed is based on a series of corrections to the ideal mesh. The results will be a reasonable approximation of stiffness since the mesh laminate is not a homogeneous isotropic material.

Consider first a perfect lamination, i. e., the lamination is of constant thickness (parallel surfaces), and the strands have rectangular cross sections and are not twisted or bent. (See Figure 8.)

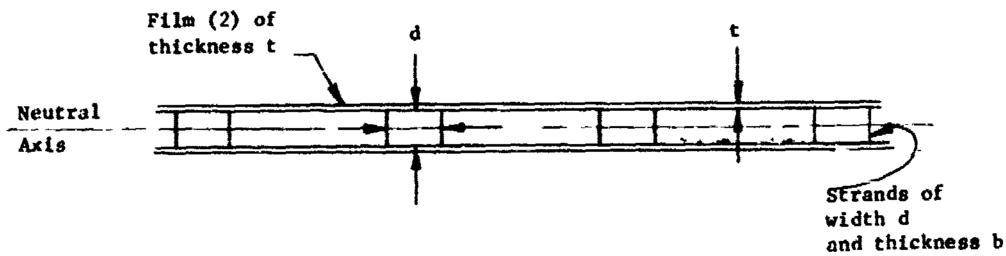


FIGURE 6
PERFECT LAMINATION

The moment of inertia of a rectangular cross section of the mesh alone is given by:

$$I = \frac{1}{12} bd^3$$

The moment of inertia of a 1" wide strip of mesh with N integral number of strands is:

$$I = N \frac{1}{12} bd^3$$

The EI product for the perfect mesh alone is then

$$E_{AL} \ (N \frac{1}{12} bd^3)$$

The moment of inertia due to the film is:

$$I = 2\ (I_o + Ax^2)$$

Where I_o = the moment of inertia of one side of the film about its own neutral axis, A = area of one side of film 1" wide, and $X = \frac{d}{2} + \frac{1}{2} t$.

The EI product for two films is therefore

$$2E_{film}(I_o + Ax^2)$$

The sum of these two terms is the EI product for the perfect mesh laminate:

$$(EI)_{total} = E_{AL}\left(N\frac{1}{12}bd^3\right) + 2E_{film}(X_o + Ax^2)$$

The actual mesh laminate differs from the configuration assumed for the perfect lamination. Three corrections are made to the expression for the $(EI)_{Total}$ product to account for the dimensional difference between actual and perfect mesh configurations.

Each strand in the mesh was assumed to be parallel, i. e., not twisted. This is not always the case; the strands assume an angular position, ϕ, within the mesh. This angular position in some cases increases and in some decreases the EI products of the mesh.

The moment of inertia with mesh strands at an angle 0 is given by:

$$I_\phi = N\left[I_x + A\left(\frac{b}{2}\tan\phi\right)^2 \cos^2\phi + I_y + A\left(\frac{b}{2}\right)^2 \sin^2\phi + P_{xy} + A\left(\frac{b}{2}\right)^2 \tan\phi \sin^2\phi\right]$$

Where N = Number of strands per inch of mesh

A = Cross Sectional Area of the strand

b = Strand Thickness

Y_x = Moment about neutral x axis of a strand

I_y = Moment about neutral y axis

ϕ = Angle that mesh deviates from ideal case

P_{xy} = Product of inertia = 0 for all mesh cases

Microscopic examination of the laminate shows the cross section of the strand to vary from point to point. These variations cause changes in the moment of inertia of the mesh. Also, many nicks exist perpendicular to the strand length which also reduce the stiffness. It is estimated that these mechanical variations reduce the mesh stiffness by 10 per cent. This arbitrary correction is introduced in the stiffness calculation.

The film is purposely laminated to a different configuration than shown as the ideal mesh. An approximation to the film is shown. Imperfections such as high points, low points, and wrinkles exist with present techniques of laminating.

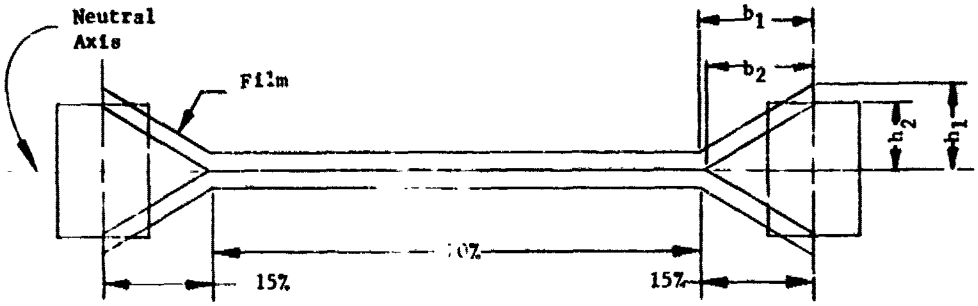


FIGURE 7
APPROXIMATE FILM CONFIGURATION

The actual EI for the films can be expressed in terms of the LWD and SWD of the mesh. The EI product in the transverse direction is:

$$\text{Transverse } EI_{film} = \frac{2}{SWD}\left[\frac{.7}{3}\, SWD\, t^3 \quad 2\,\frac{1}{12}\, b_1 b_1{}^3 - \frac{1}{12}\, b_2 b_2{}^3\right] E_{film}$$

$$\text{Transverse } EI_{film} \approx \frac{2}{SWD}\left[\frac{.7}{3}\, t^3 + 2\,\left(\frac{1}{12}\quad (.15)(SWD)+t\right]\left(\frac{d}{2} + t\right)^3\right.$$

$$\left. -\frac{.15}{12}\quad (SWD)\,\left(\frac{d}{2}\right)\right)\Bigg] E_{film}$$

Where t = film thickness

d = strand width

SWD = Short Way of Diamond

LWD = Long Way of Diamond

The EI product in the machine direction can be found with the above equation by substituting LWD for SWD.

Table II shows the calculated values of stiffness and comparative ratios. Calculations were carried out for both the transverse and machine direction for each mesh laminate and expressed in units of lb.-in^2/in.

Numerical values used for computation are:

E Mylar = 0.6×10^6 lb./in.^{2}

E Polypropylene = 0.15×10^6 lb./in.^{2}

E_{At} = 10×10^6 lb./in^2 (both 1145 and 5052 alloy)

ϕ = 30° for both # 1 and # 2 mesh in 5052 alloy and #8C in 1145 alloy, and 45° for both # 1 and # 2 mesh in 1145 alloy.

The first approach tried to experimentally determine stiffness was the cantilever beam method. A 1" x 6" strip was fixed in a horizontal position by clamping two inches on one end. The measured vertical deflection caused by its own weight could be used to determine the experimental EI product. This method was discarded because the test strip would not lay flat enough to obtain accurate measurements of deflection.

The second method used was a ring deflection method. This method consisted of cutting 1" x 6" sample strips of each type of material in the transverse and machine direction. Each sample was formed into a circle or ring of 6" circumference and joined. Each sample was hung over a straight stiff wire mounted horizontally. Another small length of wire formed the lead and was placed in the bottom of the ring 180° from the top suspension point. The weight of the load wire caused the ring to take an eliptical shape. Measurements of the major and minor diameters were made to the nearest 1/16 inch. These measurements were subtracted from the original diameter to obtain D_x and D_y, the changes in horizontal and vertical diameters, respectively. The vertical radial displacement of the ring is

$$\frac{1}{2} D_y = \frac{WR^3}{2EI} \left[\frac{1}{(\sin \alpha)^2} \left(\frac{1}{2}\alpha + \frac{1}{2} \sin\alpha \cos\alpha\right) - \frac{1}{\alpha} \right] \quad (1)$$

The horizontal radial displacement at the midpoint of the ring between leads is

$$\frac{1}{2} D_x = \frac{WR^3}{4\,EI} \left[\frac{2}{\alpha} - \frac{1}{\sin} - \frac{\alpha \cos \alpha}{(\sin \alpha)^2} \right] \quad (1)$$

Where W = Lead Weight

R = Radius of Ring

α = 1/2 the angle between the applied leads

In our case, $\alpha = \frac{\pi}{2}$; $W = 1.81 \times 10^{-3}$ lb, R = .952". Therefore, to find the EI product of our sample rings, these equations reduce to:

$$FI = \frac{2.135 \times 10^{-5} \text{ lb}}{D_x} \text{ in}^2/\text{in.}$$

$$EI \quad \frac{2.315 \times 10^{-4} \text{ lb}}{D_y} \text{ in}^2/\text{in.}$$

For each ring two calculations of EI were made, one using D_y and one using D_x. Three rings were constructed for each material giving six values for the EI product for each type of material in each direction. The six values for each EI product were averaged and recorded.

The above values are not corrected for deflection of the ring caused by its own weight. The stiffer the material the smaller will be its

(1) "Formulas for Stress and Strain," by Raymond J. Roark, page 153, case No. 9.

deflection caused by its own weight.

This curve was constructed from zero external load measurements made on sample rings. The stiffest material measured had no observable deflection under zero external load. Corrected experimental EI products are shown in Table VII.

The ring deflection method was also used to find the experimental values of the EI product of A12 material. The same experimental equations apply when the right hand side is multiplied by the factor of $(1 - u^2)$ where u = Poisson's Ratio.[2] The analytical determination method for the A12 material converted the aluminum to an equivalent in Mylar.[3]

The following equations were used for the conversion:

$$I_{A_{12}} = I_{Mylar} + I_{eff}$$

$$I_{Mylar} = \frac{1}{12} t_1^3$$

$$I_{eff} = 2 \left[b_c\, e/12 + b_c\, t \left(\frac{t + t_1}{2}\right)^2 \right]$$

$$EI_{A_{12}} = E \left[\frac{1}{12} t_1^3 + 2\left(b_c\, t/12 + b_c t \left(\frac{t + t_1}{2}\right)^2\right) \right]$$

Where t = Thickness of aluminum

t_1 = thickness of Mylar

b_c = equivalent width of aluminum

$$b_c = \frac{E_{al}}{E_{Mylar}} b_1$$

b_1 = width of sample ring

It was found that Echo I material (0.5 mil metallized Mylar) is not stiff enough to use the ring method for determination of its stiffness.

The results of all calculations and their ratios are included in Table V. A plot of analytical to experimental stiffness values is shown on Drawing Figure 10.

Both the analytical and experimental calculations show that # 1 mesh 5052 alloy gives the stiffest combination. The rest of the material combinations are closely grouped. It is suspected that not all variables have been accounted for in the analytical determination of stiffness of the mesh materials. We feel that the experimental measurements are more realistic of the actual material stiffness.

(2) "Formulas for Stress and Strain," by Raymond J. Roark, page 121, Section 36.

(3) "Mechanics of Materials, "Miller and Doeringsfeld, Section 10-6

TABLE V.
EXPANDED MESH LAMINATE RIGIDITY

Material	Lb-in^2/in Anal.	lb-in^2/in Exper.	Exper/ Anal.	EI(Anal)/ Echo I(Anal)	EI(Exp)/ Echo I(Exp)	EI(Anal)/ Al2(Anal)	EI(Exp)/ Al2(Exp)
Echo I (0.5 mil metallized Mylar)	6.25 x 10^{-6}	7.28 x10^{-6}	1.165	1.0	1.0	.025	.029
Al2 (Laminate of 0.18 Al. on each side of 0.35 mil Mylar)	253 x 10^{-6}	250 x10^{-6}	.979	41	34	1.0	1.0
Mesh Configuration							
#1 (3x6)(5052)PP Mesh, Trans. Dir.	3551 "	1565 "	.451	568	215	14.02	6.26
#1 (3x6) (5052) PP Mesh, Mach. Dir.	1540 "	495 "	.322	146	68	6.08	1.98
#1 (3x6) (5052) Mylar Mesh, Trans. Dir.	3677 "	1655 "	.451	588	227	14.52	6.62
#1 (3x6) (5052) Mylar Mesh, Mach. Dir.	1734 "	627 "	.362	277	86	6.84	2.50
#1 (3x6) (1145) PP Mesh, Trans. Dir.	2018 "	1018 "	.505	323	140	7.97	4.02
#1 (3x6) (1145) PP Mesh, Mach. Dir.	895 "	458 "	.513	143	63	3.53	1.80
#1 (3x6) (1145) Mylar Mesh, Trans. Dir.	2044 "	991 "	.484	327	136	8.07	3.91
#1 (3x6) (1145) Mylar Mesh, Mach. Dir.	1054 "	545 "	.517	169	75	4.16	2.15
#2 (2x5) (5052) PP Mesh, Trans. Dir.	1856 "	968 "	.521	297	133	7.33	3.87
#2 (2x5) (5052) PP Mesh, Mach. Dir.	1117 "	540 "	.484	179	74	4.41	2.16
#2 (2x5) (5052) Mylar Mesh, Trans. Dir.	2101 "	712 "	.339	336	98	8.29	2.84
#2 (2x5) (5052) Mylar Mesh, Mach. Dir.	1256 "	639 "	.508	201	88	4.96	2.55
#2 (3x4) (1145) PP Mesh, Trans. Dir.	1575 "	1188 "	.755	252	163	6.22	4.69
#2 (3x4) (1145) PP Mesh, Mach. Dir.	670 "	368 "	.549	107	51	2.64	1.45
#2 (3x4) (1145) Mylar Mesh, Trans. Dir.	1700 "	956 "	.557	272	131	6.71	3.77
#2 (3x4) (1145) Mylar Mesh, Mach. Dir.	811 "	389 "	.480	130	53	3.20	1.53

NOTE: #1 Mesh is .405" LWD by .215" SWD

#2 Mesh is .280" LWD by .112" SWD

V. EXPANDED MESH POTENTIAL

The potential of expanded aluminum mesh as a material for use in inflatable rigidizing radar reflective devices.

An optimized material for inflatable rigidizing radar reflective device must fulfill certain criteria.

A summary of some of the most important are as follows:

A. Light weight - Table III illustrates that the 3 mil x 6 mil mesh by itself would weigh less than the Echo I sphere material. It would be very advantageous to determine how to erect the vehicle without a plastic bladder. It would appear to be logical that progress can also be made toward reducing the material weight by reducing its original thickness. If one mil thick material could be expanded with a strand width of 3 mils then the weight shown in Table III would be reduced by a factor of six.

   Thus a 135 ft. diameter satellite would weigh only 22.5 lbs. which would be 9.7% of an equivalent sized Echo I weight, and 4.75% of the Echo II weight.

   If a plastic bladder is required it is obvious that it must be as light as possible.

B. Rigidity

   Even if a plastic bladder is necessary the mesh material has the advantage over metallized plastics of rigidity. Table V. illustrates this. This table also shows that the mesh material is more rigid than the Echo II material.

   The shape of the mesh is ideal for rigidity because of the large effect that necessary strand width has on maximizing the EI product. This can be seen in Figure 1.

C. Open Area

   Open area is very important because it reduces the effects of solar pressure and molecular drag forces. The expanded mesh can obtain open areas as high as 96% with one mil original material thickness. This would result in 1/5 the drag and buckling force which would result from 90% open material.

D. Long Space Life

   No significant deterioration due to space environment can be envisioned for the expanded aluminum mesh.

E. Fabrication Capability

   The expanded mesh material is available and has been experimentally used in spheres up to 12½ feet in diameter. Folding and packaging tests etc. have been performed. No major fabrication developmental problems remain unsolved.

EXPANDABLE FOAM-IN-PLACE SHELTERS AND RELATED ITEMS

Jack F. Furrer

U. S. Army Natick Laboratories

While the work on expandable structures to be presented here does not deal with aerospace application, I believe that it is of sufficient interest to warrant your consideration for potential application in this area.

The U. S. Army Natick Laboratories became heavily involved in foam plastic development in 1955 when it entered into a program to develop foam plastic energy dissipators for Aerial Delivery application. In 1956, it became apparent that the high strength polyurethane foams also had potential as a structural insulating material for shelters. Starting at that time and continuing until fairly recently, a number of shelters of varying sizes and configurations have been producted by conventional spray equipment. The purpose of this work was to establish the design criteria necessary for shelters to be produced by simpler means. One of the results of this work was a preliminary stress analysis of foam plastic shelters, which is utilized to estimate the basic physical and mechanical parameters for various sized shelters.

It was realized early in the program that, unless the process was extremely simple and foolproof, its utilization would be severely restricted. A thorough evaluation of the problems led to the formulation of the concept of a self-propagating solid foaming sheet. This was visualized as a solid resinous system which was composed of finely divided particles bonded together to form a flexible sheet and fabricated in the final shelter foam. This formed sheet would be inflated by air or by gases from the chemical reaction. Heat applied in one area would initiate the reaction and self-propagation would result in rapid foaming and setting within a short period of time.

Encapsulation of the active components was considered and rejected because of danger of accidental triggering, cost, and inability to handle material such as TDI. Effort was then devoted to the new area of blocked isocyanate adducts.

A contract with General Mills resulted in a number of moderately stable solid compositions; however, the heat required for unblocking and reaction initiation was far in excess of that available in the field.

At this point, the program funding was picked up by Canada under the U.S.-Canadian Development Sharing Program. A contract was negotiated with Ontario Research Foundation to establish the feasibility of a foaming sheet.

Initially, attention was concentrated on the continuation of work started by General Mills since urethane foam technology was well advanced, and they produced foams with desirable properties. Again, however, attempts to obtain foamable compositions that were both exothermic and storage stable at normal ambient temperatures were unsuccessful. At this point, a thermodynamic study convinced us that it was not possible to obtain highly exothermic reactions

from this class of materials. Combinations of isocyanate adducts and polyepoxides, which theoretically should be highly exothermic, also yielded no useable foam compositions.

Attention was then directed toward systems based upon epoxide amine condensation reactions, and a number of compositions were developed which were highly exothermic. Initiation temperatures range from 160°F. up, and peak exotherms as high as 500°F were recorded.

In general, these formulations are composed entirely of solid components in the form of free flowing, finely-divided powders. They yield rigid foams with essentially closed cell structure in density ranges of 2-5 lbs/cu ft. The compressive strengths are somewhat below those for urethanes of the same density. Some of these formulations which have been subjected to room temperature storage were found to be stable in excess of six months. They have initiation temperatures of 200-300°F and are self-sustaining when heated to these temperatures.

Even with the most exothermic epoxy formulations, it was found that the objective of a system that would self-propagate as a thin layer could not be attained. Consequently, an auxiliary heating source has been developed to provide the heat necessary to melt the components of the foamable composition and raise the temperature to that required to initiate the foaming reactions. Because of their ease of control, pyrotechnic materials were examined for this purpose, and the most suitable composition was shown to be that based on the oxidation of iron powder with elemental sulphur. It has been established that flexible sheets can be obtained from powdered pyrotechnic mixtures by compounding with a small percentage of loose asbestos fibers. Although of low strength, the sheets show no tendency to dust even with as little as 2% of the fibrous bonding agent, and the strength can be improved by the addition of glass fibers. Optimum burning properties are obtained with pyrotechnic sheets containing iron and sulphur in ratios from 75:25 to 65:35 by weight and bound with 2-10% asbestos. The sheets are self-propagating above a total weight of 0.1 lbs/sq ft and have an average burning rate of 1 ft/min. They require only a simple ignition device, such as a match.

The concept of a self-contained expandable sheet has been demonstrated by combining this heat source with a suitably packaged epoxy foam formulation. For testing, the powdered formulation was packaged in polyethylene coated aluminum foil in the form of a channelled bag. The heating sheets were bonded to the outer surfaces of the aluminum envelope with an insulation layer interposed to decrease the peak temperature transmitted to the foamable composition. This procedure for producing a flexible unit was adopted for laboratory convenience and does not necessarily represent the most suitable design for manufacture.

With composite sheets of this description, employing either a single pyrotechnic sheet on one surface or pyrotechnic sheets sealed to both surfaces, acceptable foam products have been obtained at ambient temperatures in the range -40°F to 85°F. The occurrence of large pores in the foam was encountered and attributed to entrapment of air in the aluminum envelope, since this effect was not observed in oven foaming experiments.

At the lowest temperature at which tests were performed, -40°F, it was

shown that a pyrotechnic content of 45-55% by weight of the foamable layer is required to give satisfactory volume expansion, whereas at 85°F approximately 30-40% of the pyrotechnic mixture is sufficient. Although a detailed investigation has not been made of the operable temperature range for a given ratio of pyrotechnic to foamable components, an indication of the flexibility of the system is given by the fact that suitable products resulted from tests on identical composite sheets at temperatures of 23°F and 77°F.

To illustrate the use of the composite sheet in the form of an actual shelter, a model structure consisting of a three-foot diameter hemisphere was prepared in which a single, inner heating sheet was employed corresponding to 32% by weight of the foamable layer. On testing at an ambient temperature of 86°F, a rigid and essentially uniform foamed structure was obtained with average wall thickness of 2 inches. Thus, the concept of a foamable sheet and its applicability for shelter construction have been clearly established. In addition, the potential of the system for operation under arctic conditions has been demonstrated.

Work is expected to continue in this area to optimize the formulation, investigate the most feasible methods of packaging, and analyze problems of converting continuous sheet material to various shelter configurations.

Concurrent with this program is one to develop a self-contained kit to produce a definite amount of foam. This kit would be utilized by the individual soldier under ambient field conditions

Two approaches are currently under active investigation, both with Atlantic Research Corporation. The first is the development of a simple, inexpensive heat gun utilizing a readily available fuel such as gasoline. Into the flame is fed one or more solid foam compositions. The present crude prototype has given encouraging results. Using unexpanded styrene beads coated with an epoxy resin, foams having fair physical properties have been produced. Retention on a vertical wall exceeds 90%.

The other approach is directed toward an all-liquid aerosol type spray kit which will function over the entire range of climatic temperatures. Essentially, such an approach requires liquids that maintain a low viscosity from -65 to 140°F. In the early work on energy dissipators, some consideration was given to extremely low temperature foaming reactions. Out of this limited effort, evolved the possibility of producing foams from ionicly catalyzed vinyl monomers. Admittedly, the foams were of poor quality, but the basic monomer had low viscosities at -40°F and the reactions proceeded rapidly. For the past year, we have been aggressively pursuing this approach for the kit. To date it appears that vinyl ethers offer the greatest chances of success. While foams have been produced from these ethers, their physical properties to date have been poor, and much work remains to be done on control of the relative rates of polymerization action and foaming and on the relation of physical properties of the foams to monomer composition.

AIR-SUPPORTED TENTS FOR MILITARY USE

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Modern scientific and technological developments made in military equipment and in support of a mobile army have resulted in the need for new type tentage. The need for the new tentage varies from highly specialized items for the missile program to large maintenance tents for ground vehicles and aircraft. Operational requirements for tents that most effectively support mobility concepts can be found in references 18 and 19, and may be summarized as follows:

Maximum utility space - To provide necessary work areas, unhampered by internal supporting poles and frames.

Lightweight, low bulk and cube - To increase transportability.

Minor maintenance and site requirements - To minimize logistical support.

Expandability - To reduce number and types.

Versatility and adaptability - To meet unique military field tentage requirements.

Easy and quick erecting and striking capability - To conserve time and manpower.

Protection capabilities - To protect field combat troops against environmental stresses and CB agents.

The use of air-supported tents, which can be easily transported, erected, and struck for more mobile Army operations, represents one approach taken by the Army to provide mobile shelters of reduced weight, cost, and cubage.

With the development of air-supported tents, the technology of tent making is developing step by step from a traditional craft to a branch of scientific engineering. This is a parallel development: Production engineering and scientific design. These two are interdependent for progress and support; the tents cannot be fabricated without materials--materials cannot be developed without knowing the end use conditions of the tent.

Development engineering is defined as the building of prototypes meeting user requirements. Scientific design is defined as the design of the tent--the starting point of development engineering--including research and development on structures and materials which will produce an end item suitable to meet the operating characteristics required for Army operations.

It is the intent of this paper to present both development engineering and the scientific design of air-supported Army tents. The development engineering phase will be covered by a brief discussion of some of the air-supported tents developed for the military. The scientific design phase will be covered by discussing some of the interesting aspects of the work leading to the development of air-supported tents, and pointing out areas which require additional research and development.

PRODUCTION DESIGN FOR AIR-SUPPORTED TENTS

In this portion of the report, some of the air-supported tents developed for the military will be described along with information relative to background, unique features, and pertinent test data. The development and testing of the air-supported tents can logically be divided in three parts:

(1) An exploratory study of the development and testing of air-inflatable structures.

(2) Development of special-purpose tents in support of the missile program--single-wall, air-supported tents.

(3) Development of maintenance tents--double-wall, air-supported tents for Army vehicles and missiles.

Under the exploratory study, two tents will be discussed: A one-man mountain tent developed for the Army and a 20 x 40 inflatable air-matt maintenance tent developed for the Air Force.

A. One-Man Mountain Tent (Inflatable)

This tent was developed to meet the need for a self-contained unit which could be easily and quickly erected under extreme cold and high winds. The tent (Fig 1) developed for the Army (21) measures 2 feet 11 inches wide, 7 feet 7 inches long, 3 feet 2 inches high in front decreasing to a width of 1 foot 9 inches, and a height of 1 foot 4 inches in the rear. It weighs 14 pounds. The tent consists of an A-shaped frame made from three inflatable tubes running down the front and across the ridge. The tent was tested at Mount Washington during the winter of 1949-50. As a result of tests, the tent was found to be too heavy and bulky for its intended use. In addition, after two weeks of use in cold weather, the inflated beams developed minute air leaks at almost every seam and corner, requiring frequent reinflation to keep the tent erect.

Fig 1 - One-Man Mountain Tent (Inflatable)

Fig 2 - Shelter, Inflatable, 20' x 40'

B. Shelter, Inflatable, 20 x 40

This tent was developed by the Air Force as a maintenance shelter (15). The tent (Fig 2) was constructed of air-matt material 3 inches thick when inflated. It covered an area 20 feet wide by 40 feet long. The tent was tested at Fort Churchill, Manitoba, Canada, during the winter of 1953-54 and Mount Washington, New Hampshire, during the winter of 1954-55. This tent, when compared to the same size standard frame-type tent, was lighter in weight, more compact, and easier to transport and erect. However, the tent required more fuel to heat than an insulated frame-type tent and was not considered sufficiently durable for field operations in cold weather. The air-matt material from which the tent was made became too stiff and developed minute air leaks which were difficult and in some instances impossible to locate and repair.

The two field trials on air-inflatable tents provided the following information:

a. The full benefit of air-supported structures cannot be realized in small personnel tentage--one- and two-man tents; it can only be realized in larger maintenance and special-purpose tentage needed to meet specific operational requirements.

b. The tents tested lacked the durability required for Army field operations. Nevertheless, it was considered that the principle inherent with air-supported tents; i.e., low weight, bulk and ease of erection, possessed sufficient merit to warrant further development. Accordingly, the Army entered into the second phase of its development of air-supported structures.

The second phase consisted of the development of single-wall tents to meet specific Army requirements in support of the missile program. These tents possessed unique operational features difficult to obtain by other means of construction and certainly could not be met with conventional tentage of the pin, pole or frame type. One feature is the quick-release mechanism, which provides the means for instant striking of the tent. The quick-release device (Fig 3 and 4) utilizes a heavy-duty slide fastener and lanyard. A quick pull on the lanyard opens the tent in halves, taking advantage of the inflation pressure to throw the sides back and away from the equipment. A second feature is radar transparency of the thin skin structure, which is especially important for shelters covering tracking radar.

Missile shelters are required to protect the missiles and sensitive electronic radar equipment against the damaging effect of wind, rain, and snow; also, to shelter maintenance crews performing routine check-out services as well as maintenance operations.

Six tents, based on the single-wall, air-support principle, will be discussed. These are as follows: Shelter, Air-Supported, Vertical Check-Out (Redstone); Tent, Nike Hercules, Air-Supported; Tent, Air-Supported,

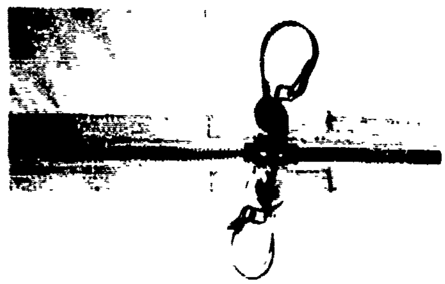

Fig 3 - Quick Release and Slide Fastener (Closed)

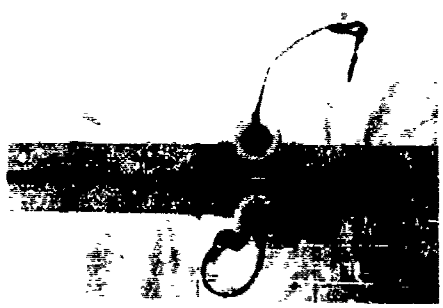

Fig 4 - Quick Release and Slide Fastener (Opened)

Nike Hercules Launch Area for Field Army; Tent, Air-Supported Launcher, Hawk System; Tent, Radome for Track Antenna and Acquisition Radar Sets; Tent, Air-Supported, Storage Type.

C. Shelter, Air-Supported, Vertical Check-Out (Redstone)

This tent was developed to provide all-weather protection for the missile and personnel performing check-out operations (10, 18). The shelter (Fig 5) measures 24 feet in diameter and stands 18 feet 6 inches high. This tent met all the requirements for attachment and rapid erection and removal, and provided full protection against weather including winds up to 40 mph. It is so designed that it can be attached while the missile is horizontal, raised to a vertical position, and inflated within 4 minutes. The shelter can be released from the missile by pulling a cord which disconnects the shelter and allows it to drop to the ground away from the launching pad in two to three seconds.

This development constituted an exploratory effort in an area never before attempted--the enclosure of a missile thrust unit within an air-supported structure. The scientific and technical gains have contributed to the successful development of the other air-supported tents which follow.

D. Tent, Nike Hercules, Air-Supported

This tent (Fig 6) was developed to provide all-weather protection for the missile on the launcher and to shelter personnel performing check-out and maintenance services on the missile and launcher (18, 19). The tent measures 17 feet 6 inches in diameter and is 61 feet long and 13 feet high. The tent is designed to be installed on a recessed base, located between abutments, so that it will be partially protected from the full impact of the wind.

This tent is equipped with a quick-release device which is manually activated by pulling a lanyard. A quick pull of the lanyard allows the shelter to split in two halves along its entire length, falling away from the missile which it does in 1 to 2 seconds.

E. Tent, Air-Supported, Nike Hercules Launch Area, for Field Army

This tent was developed to provide protection to personnel while performing maintenance and check-out services on the missiles on the launcher and to provide space for two additional missiles on the racks (19). This tent (Fig 7) measures 20 feet wide, 70 feet long, and 20 feet high. The tent is equipped with a manually activated quick-release device which permits striking the tent in approximately 3 seconds. The development on this tent is completed and prototype tents are being fabricated.

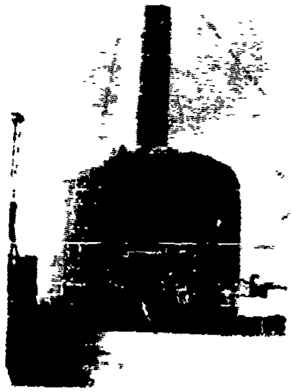

Fig 5 - Shelter, Air-Supported, Vertical Checkout (Redstone)

Fig 6 - Tent, Nike Hercules, Air-Supported

F. Tent, Air-Supported, Launcher, Hawk System

This tent (Fig 8) was developed to provide cold climate protection to the Hawk Missile on the launcher and to operating personnel performing check-out and maintenance services (19).

This tent is a 3/4 sphere, 28 feet in diameter and 21 feet 6 inches high. The tent consists of an outside envelope and an inside shell with an insulating air space between. The tent is erected over the Hawk launcher and the missile is fired directly through the tent.

G. Tent, Radome, Air-Supported for Track Antenna and Acquisition Radar Sets

This tent (Fig 9) was developed to provide all-weather protection for both acquisition and tracking antenna. The tent is a 3/4 sphere, 27 feet in diameter and approximately 20 feet high (18). It is adaptable to both tower and ground mounts on a specially prepared base. The radome is equipped with a quick-release fastener which is manually activated by pulling a lanyard, which divides the tent in two halves completely uncovering the antenna. This feature was felt necessary in the event of operational interference caused by certain environmental conditions, such as the sudden formation of ice, or the radome skin being wet from dew or rain. This device makes it possible to remove the radome quickly without interrupting sensitive tracking operations.

H. Tent, Air-Supported, Storage Type

This tent (Fig 10) was procured because of the Army's interest in using air-supported tents for warehouses (18). The tent measured 40 feet wide, 80 feet long, and 15 feet high.

The tents were erected and tested in the New England area for extensive periods. They have demonstrated their capacity to withstand wind, snow, and degradation from the elements. Even though the tents were developed as an air-supported structure. this tent, when deflated, may be utilized as a tarpaulin to protect critical supplies and materiel in outside storage areas. It can be quickly inflated by blowers so that individuals and materials handling equipment will have access to supplies when necessary.

Tents based on the single-wall, air-support principle, while successful in meeting special military requirements, have an obvious drawback for use as maintenance tents. This drawback is the need for special doors and air locks to permit entry and exit of large vehicles for maintenance work. Accordingly, concurrent with the development of single-wall tents, the Army entered into the third phase of the development of double-wall, air-supported tents. The advantages of these tents for maintenance work are obvious. The supporting air is contained between

Fig 7 - Tent, Air-Supported, Nike Hercules Launch Area, for Field Army

Fig 8 - Tent, Air-Supported, Launcher, Hawk System

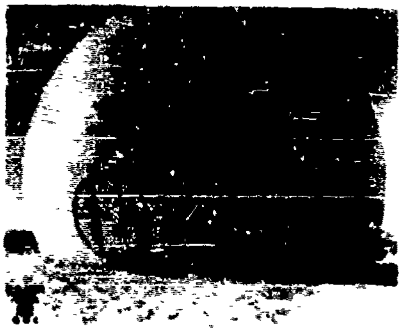

Fig 9 - Tent, Radome, Air-Supported, for Track Antenna and Acquisition Radar Sets

Fig 10 - Tent, Air-Supported, Storage Type

two walls, forming the sides and roof of the tent, leaving wide open ends for entry of vehicles. Three tents of this type will be discussed: (1) Tent Set, Vehicle Maintenance, Small, Air-Supported for Arctic Use; (2) Tent, Maintenance, Multi-Purpose, Inflatable, Sectionalized; and (3) Tent, Air-Supported, Double-Wall, Assembly Area, Nike Hercules, Mobile System.

I. Tent Set, Vehicle Maintenance, Small, Air-Supported for Arctic Use

This tent (Fig 11) was developed as a portable tent to be carried by maintenance personnel to a disabled vehicle in the field (19). The dimensions of the tent are 20 feet wide, 13 feet long, and 12 feet 6 inches high. The sides and roof of the tent are double-wall construction with the walls 20 inches thick. The end curtains are single wall. The tent is of the nose-in maintenance type. It provides protection to the maintenance crews in cold climates. The tent is equipped with fitted end curtains for each of the vehicles it will service.

J. Tent, Maintenance, Multi Purpose, Air-Supported, Sectionalized

This tent (Fig 12) was developed to cover the entire Pershing Missile during maintenance and check out under cold weather conditions (19). The dimensions of the full size tent are 20 feet wide, 52 feet long, and 12 feet 6 inches high. The tent consists of four units of the Tent Set, Vehicle Maintenance, Small (Fig 11), butted and fastened together. This tent has been made a component of the Pershing Missile System. Shelters are currently being procured for the Pershing System.

K. Tent, Air-Supported, Double-Wall, Assembly Area, Nike Hercules Mobile System

The tent (Fig 13) was developed to house the Nike Hercules missile and missile components, and to shelter personnel while performing the war heading and assembly of missiles (19). The dimensions are 48 feet wide, 72 feet long, and 24 feet high. The tent consists of six 12-foot sections joined with catenary and collar fasteners which permits personnel to join the section in cold weather while wearing Arctic clothing. The sections are hemi-cylindrical in shape and double-wall construction with the walls three feet thick. The end curtains are of single-wall construction. Each end curtain contains personnel and utility doors. This tent has been standardized as a component of the Nike Hercules Mobile System.

With the development of the Nike Hercules Assembly Area Tent, it was found that double-wall tents, 48 feet wide, are practical. The Army is currently considering the feasibility of extending the width of the double-wall tents to 110 feet for use as aircraft maintenance hangars when a requirement for this type shelter is established.

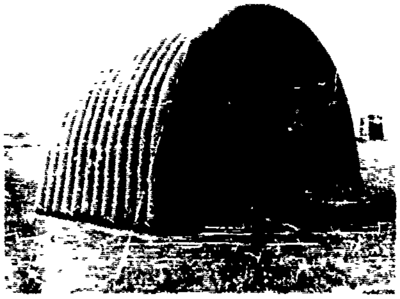

Fig 11 - Tent, Set, Vehicle, Maintenance, Small, Air-Supported for Arctic Use

Fig 12 - Tent, Maintenance, Multi-Purpose, Air-Supported, Sectionalized

Fig 13 - Tent, Air-Supported, Double-Wall, Assembly Area Nike Hercules Mobile System

SCIENTIFIC DESIGN OF MATERIAL FOR AIR-SUPPORTED TENTS

In completing the description of the air-supported tents developed for Army use, the production engineering phase is concluded. Naturally, one is curious to know more about the factors which were considered in the development of these tents. The information in support of production engineering is scientific design. It is in this area that observations were made which is felt should be shared with all agencies interested in the development of air-supported tents. These observations are limited to the following investigations:

a. Effect of shape on the aerodynamic behavior of a tent.

b. Fabrics used by the Army.

c. Discussion of some fabric properties considered important in the development of air-supported tents; i.e., weight, flexibility at low temperatures, and dielectric constant.

As noted, the air-supported tents developed for the Army varied in shape and size from spherical radome to cover radar antenna to cylindrical tents with rounded or flat ends to cover missiles or for use as maintenance tents. Information on the design of these shelters was obtained from available sources (2, 3, 4, 5, and 8). While the spherically-shaped radome has been subjected to aerodynamic testing from which a design manual has evolved (4), no such progress has been made for other shapes. Information for the design of Army air-supported tents of other than spherical structures was assembled from engineering data obtained from other sources (3, 5, and 8).

To obtain aerodynamic data on flexible, cylindrical, air-supported tents, a limited wind tunnel study was conducted at Massachusetts Institute of Technology. The model selected for test was a 1/10th scale Tent, Nike Hercules, Air-Supported. The interesting aspects of the report of test can be summarized here.

The first of these was the development of scaling parameters (2). It was found in planning the test that the design of a flexible model air-supported tent is more complicated than a normal rigid wind tunnel model, whose shape is stable. To obtain aerodynamic and dynamic similarity between full-scale and model tents, the following parameters had to be kept the same:

Geometric shape - no wind

$$\text{Inflation parameter} = \frac{\text{Inflation Pressure}}{\text{Stream Dynamic Pressure}}$$

$$\text{Mach Number} = \frac{\text{Test Velocity}}{\text{Velocity of Sound}}$$

$$\text{Reynolds Number} = \frac{\text{Diameter x Velocity}}{\text{Kinematic Viscosity}}$$

$$\text{Aeroelastic parameter} = \frac{\text{Diameter x Fabric Elongation x Dynamic Pressure}}{\text{Fabric Stress}}$$

$$\text{Dynamic parameter} = \frac{\text{Mass of Tent}}{\text{Air Density x (Diameter)}^3}$$

Of the above parameters the ones which were considered of paramount importance were: Shape, aeroelastic parameter, which is related to the stiffness of the fabric, and the dynamic parameter, which is related to fabric weight. From experience, it was known that Mach Number effects are small for cylindrical bodies with rounded ends below values of 0.25 or 150 miles per hour at sea level in air.

The Reynolds Number determines the flow pattern as it is influenced by viscous effects. Again, from experimental work, the major variation of flow usually occurs below certain values of this parameter. If one stays well above such a value, little change in the flow pattern will take place over a large variation of the parameter, and lends considerable weight in the application of the model results to full scale.

The inflation parameter is important for tent stability. From literature on air-supported radomes (4), this value is usually taken as 1; i.e., inflation pressure equal to the dynamic pressure of the air stream. In the following discussion, this value will be designated as "q".

The second interesting aspect of this study was the tent motion in the air stream. At inflation pressures below one dynamic pressure or "q", the air stream led to large deflection. For a quartering wind (Fig 14--tent, no wind), and an inflation pressure of 1 "q", the lateral deflection of the midregion of the upstream side was about 10% of the tent diameter; for 2/3 "q", the tent flapped and the deflection was about doubled (Fig 15). At an angle of 45 degrees, and with a 105-mile per hour wind, the tent quivered at inflation pressures of 5/4 "q" and 1 "q", but shook violently at 2/3 "q".

Fig 14 - Wind Tunnel Test, No Wind

Fig 15 - Wind Tunnel Test - Inflation Pressure 2/3 q, Quartering Wind

The importance of wind direction on the aerodynamic behavior of the tent cannot be emphasized too strongly. With the wind approach broadside, it was necessary to overinflate the tent 5/4 "q" to maintain stability. At 1 "q", the tent flapped violently in a periodic manner and the test could not be run at 2/3 "q". With the wind approaching the tent at angles of 135 degrees and 180 degrees, no serious motions were apparent even at the low internal pressure of 2/3 "q".

The initial wind tunnel tests were run without a zipper in the model tent. Tests were rerun with a quick-release zipper installed in the tent. The zipper installation decreased the violence of the tent motions considerably. The tent could be run broadside to the wind at 85 miles per hour with an inflation pressure of 2/3 "q". Whether the air leakage through the zipper, the change in the elastic properties of the tent, or the change in inertial properties, or some combination, is responsible for the improvement is not known. The air leakage was larger than a scaled value should be because of the model zipper size. What effect this would have is also unknown.

A review of the motion picture film taken for various test conditions and tent configurations leave the viewer with a deep appreciation of the aerodynamic effects and the necessity for making allowances in the tent design to account for these effects. From these pictures, it may be concluded that inflation pressures below 1 "q" lead to serious deflections, shaking, and transient denting of the tent. In most cases, an increase in inflation pressure to 1 "q" decreases the aerodynamic effects from violent to perhaps tolerable magnitudes. Thus, the only means at hand to reduce tent vibration and flapping is to increase the internal pressure. However, work in this area has indicated that the use of spoilers or other surface modifications of the tent will improve its operational characteristics. Further research work in this area should be expanded.

The third aspect of the study is the variations in aerodynamic and anchor loads experienced with the winds approaching the tent from different angles. As expected, the lowest lift or drag loads occur with the bow or stem end facing the wind. With the bow or stem facing the wind, the over-all lift coefficient averages approximately .15 and the over-all drag coefficient approximately .06. The over-all lift coefficient increases to .75 for wind striking the tent at 135 degrees, while the over-all drag coefficient increases to .65 with the wind striking the tent broadside. Here the aerodynamic forces show a five-fold increase in lift, and nearly double this in drag, merely from the standpoint of the wind direction. How this increase in load is being dissipated by the fabric in a tent of this shape should be further investigated.

This change in loading due to the direction of wind approach is also found in the load distribution on the anchors around the tent. Fig 16 shows the pattern of anchor loads found with the bow of the tent facing the wind. The "X" on the chart represents the anchor loads resulting from internal pressure only; the circles represent the combined load due to internal pressure and 85 mph wind. It should be noted that only those anchors around the bow and stem end appear to show an increase in load.

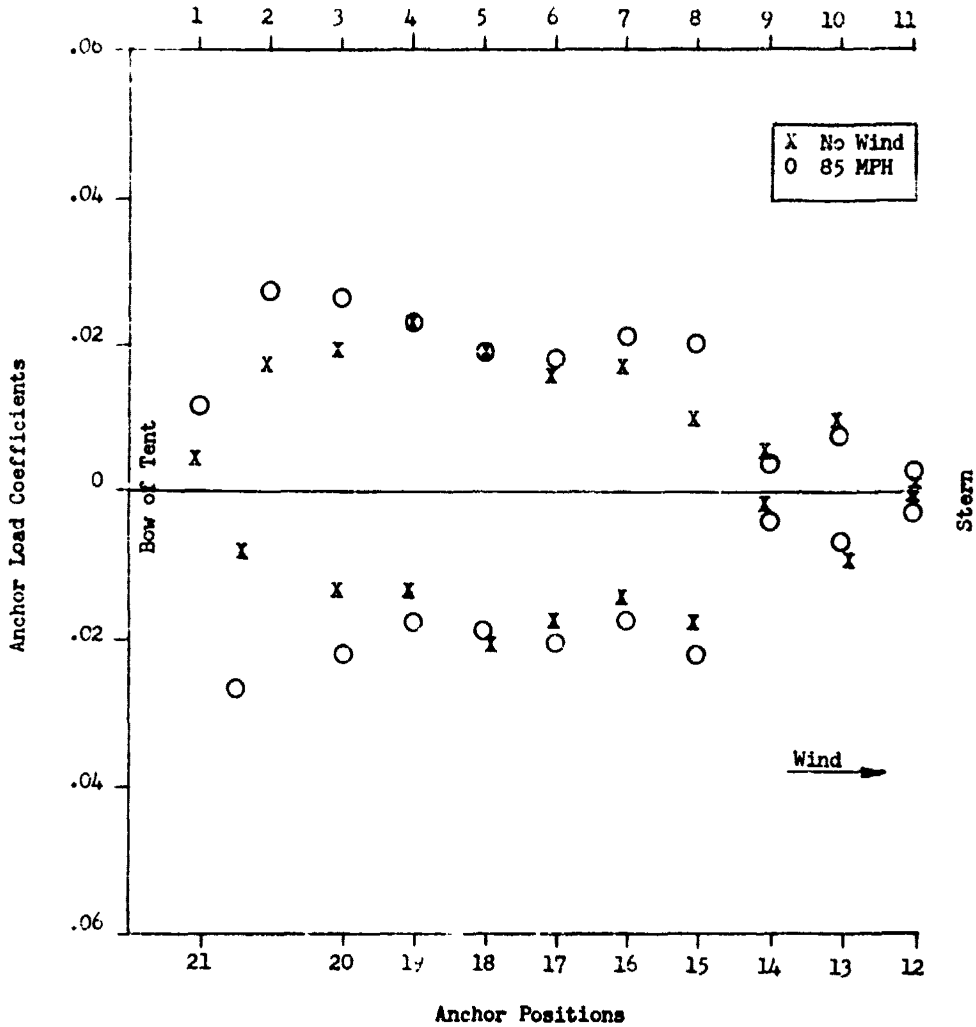

Fig 16 - Anchor Loads

Fig 17 shows the pattern found with the wind approaching the tent at 135 degrees. As expected, the pattern of anchor load distributed is different in both location and magnitude.

From the standpoint of design, it was found that for this shape tent, the maximum anchor loads are about twice the average value found from the sum of inflation and lift loads. Investigation on the anchor loads for tents of other shapes is warranted. It was concluded from the above that the aerodynamic behavior of cylindrically shaped tents is sufficiently different from those reported for radomes (4) and that design information on tents of other shapes was required. Accordingly, a new study was implemented to obtain information on a theoretical basis for the design of flexible air-supported tents of the shapes shown on Fig. 18.

It is hoped that the tent shapes proposed for test will permit extrapolation of test data not only for height of the shelter; i.e., 3/8, 1/2, or 3/4 sphere or cylinder, but for length on the basis of length-to-width ratio. The new study contains a requirement for the development of fabric stress data which was not included in the Massachusetts Institute of Technology work. Hopefully, the design data developed in this new program will establish parameters for fabric stress which, in turn, may be used as a basis for engineering fabrics to meet the specific requirements for air-supported tents. Fabrics of efficient design should result in improved shelter performance at minimum weight and cost.

The fabrics used in the fabrication of the Army air-supported tents are shown in Table I.

TABLE I

Weight, Strength, and Flexibility of Fabrics for Air-Supported Tents

		IP/DES S-62-2 Type I	IP/DES S-62-2 Type II	MIL-C-43086	Heat-Set Polyester
Fiber		Polyester	Nylon	Nylon	Polyester
Coating		Neoprene and CP*		Vinyl	None
Weight oz/yd^2		14.0	14.5	19.0	9.0
Break Strgth lbs/sq in	W	160	275	300	475
	F	160	275	300	560
Low Temp Rqr		-40 F	-40 F	0 F	-65 F
Dielectric Constant		4.1	2.8	2.8	2.7

* Chlorosulphonated Polyethylene

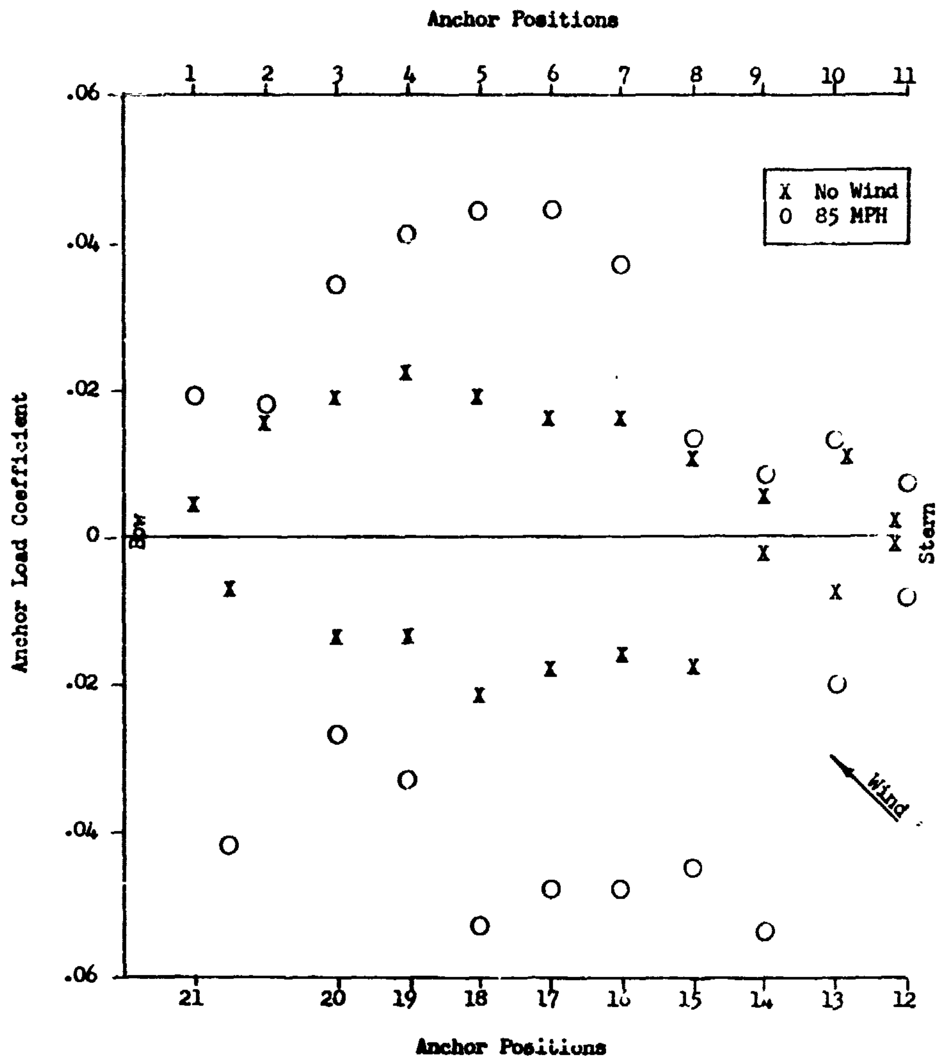

Fig 17 - Anchor Loads

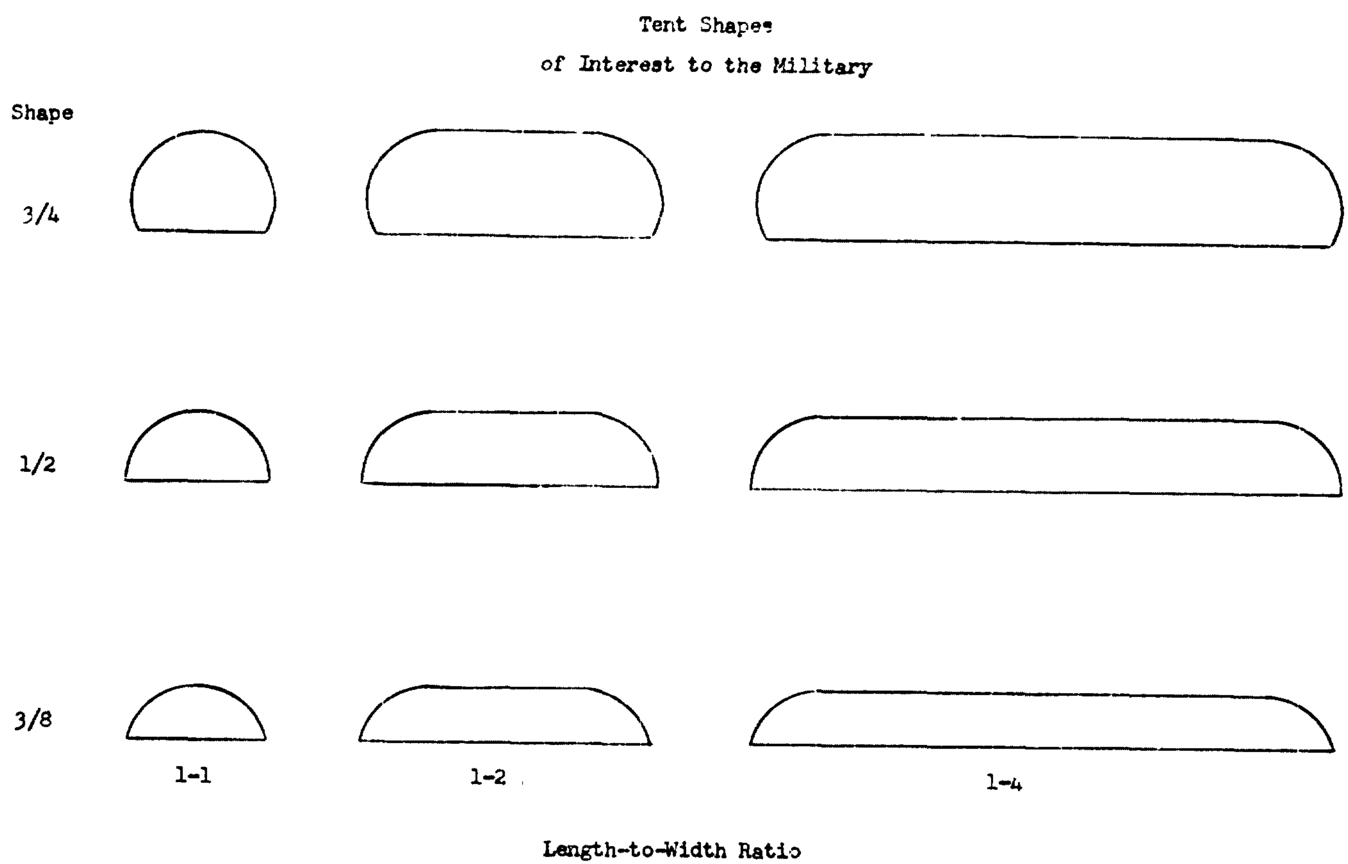

Fig 18 - Tent Shapes for Wind Tunnel Tests

The fabrics covered by IP/DES S-62-2 were developed for Arctic use where flexibility at low temperature is important. Type I polyester fabric would be used for the smaller, air-supported tents--Shelter Set, Vehicle Maintenance, Small, Air-Inflated for Arctic Use; Tent, Maintenance, Multi-Purpose, Inflatable, Sectionalized; and Shelter, Air-Supported, Vertical Check-Out (Redstone). Type II is used for the larger tents where greater strength of material is required; i.e., Tent, Air-Supported, Double-Wall, Assembly Area, Nike Hercules Mobile System, and Tent, Air-Supported Nike Hercules Launch Area for Field Army.

The vinyl-coated nylon fabric, MIL-C-43086, was developed for tents used in temperate climates where low temperature flexibility is not critical. The tents in which this material will be used are Tent, Nike Hercules, Air-Supported; Tent, Air-Supported. Radome for Track Antenna and Acquisition Radar Sets; and Tent, Air-Supported, Storage-Type.

The heat-set polyester fabric is an uncoated fabric having extremely low porosity--2 cubic feet of air per minute per square foot at six inches of water pressure. This fabric was developed primarily to improve the flexibility of air-supported tents at low temperatures. However, by eliminating the coating finish, a reduction in the weight fabric is also realized. This fabric has been developed and successfully used in single-wall, air-supported tents. A Tent, Nike Hercules, Air-Supported, made from this fabric, has been erected and struck 70 times and has been in continuous service on outdoor exposure for 18 months. The tent is still in excellent condition. The development of a heat-set polyester fabric of lower porosity is continuing so that its use can be extended to double-wall, air-supported tents, where "0" porosity at 1 psi and higher is desired.

The fabric properties listed in Table I are considered very important and warrant further discussion.

(1) Fiber Type - Early studies conducted for the Air Force (4) have shown that nylon and polyester fibers were more satisfactory than fiberglas acrylic, modacrylic and cellulosic-type fibers. Field experience with nylon radomes has been excellent. Experience with shelters made from high-tenacity polyester fibers have also proven satisfactory. Both fibers have high strength-to-weight ratios; the two fibers can be used to produce thin, relatively flat fabrics of high strength. A thin fabric of high strength is desirable, as the thickness of the fabric determines the amount of coating compound required to fill the interstices and protect the fabric. The thicker the fabric, the more coating compound, the greater the weight. Thin, lightweight, durable fabrics are prime objectives to attain in the development of fabrics for air-supported tents.

(2) Low Temperature Flexibility - The military requirements for tents to be operational at -65 F is a difficult requirement for coated fabrics to meet. It should be noted in Table I that the coated fabrics specified in IP/DES S-62-2 are not recommended for use below -40 F. In service, the air-supported tents remain relatively rigid and are subjected

to little flexing. However, they must be folded and packed for storage or shipment. The tents are not flat, but spherical and cylindrical in shape. The tents cannot be laid out flat and must be bundled into a badly creased and folded unit. The material must be capable of withstanding such handling without loss in strength. Flexibility is essential at low temperatures for ease in erecting or striking the tents under Arctic conditions.

Because of the stiffening of coated fabrics at temperatures of -40 F, the Army is continuing its development of heat-set polyester fabric which is flexible at -65 F. To develop a tent with a minimum of two years service life, it was necessary to use the 9 oz/sq yd, heat-set polyester fabric shown in Table I. If a fiber equal in tenacity to polyester fiber with better weathering resistance could be found, the weight could be reduced to as much as one-half this weight. The above indicates two areas of research to improve the flexibility of fabrics at -65 F. One of these is the development of durable fabric coating compounds which are flexible at -65 F; the other is the development of high-tenacity fibers with improved weathering resistance.

(3) Electrical Properties of Fabrics - In cases where the radome is developed to cover radar antenna, transparency to the radar signal is required. In previous studies, (4), it has been stated that good RF transmission is obtained on air-supported shelters by keeping the thickness of the material small in comparison with the wave length. The detailed design specifications for radomes generally require a minimum one-way transmission of 90% at angles of incidence up to approximately 50. In order to meet this requirement, the thickness of fabric and coating material must be kept to a minimum and the dielectric constant of the coated fabric must be low.

In the development of the fabric for Nike Hercules Radomes used in Continental United States, the work accomplished with the Bell Telephone Laboratories, Whippany, New Jersey (9) revealed that the following electrical characteristics of fabrics would effect tracking radar signals and had to be kept at a minimum value: Dielectric constant, reflection coefficient, reflection loss, dissipative loss, and boresight effect (angular shift of the radar beam).

The assistance given to the problem by the Wheeler Laboratories, Great Neck, L. I., provided a set of empirical equations which showed the correlation found among the above parameters for reflection and dissipative losses, and the dielectric constant and fabric thickness.

To determine how the desirable electrical characteristics can be incorporated into a fabric development program for air-supported tents, it is necessary to measure the dielectric constant.

(4) Dielectric Constant - This parameter is defined as a constant factor (k) which is part of an equation which describes a property of the material related to its ability to conduct lines of force of an electromagnetic field (radar signals). This is in distinction to its

electrical insulating property which is related to its ability to conduct electric current. The value of (k) may be determined from the measurement of (a) electrical capacities, (b) mechanical forces between charged conductors, or (c) wave lengths of electrical waves. All three methods yield the same result, except insofar as the value of (k) varies with the frequency of the electrical waves. The dielectric constant of air at room temperature is very nearly 1.0. For good radar transmission, it is desirable to keep the value for the dielectric constant of radome material as low as possible, approaching the value for that of air.

The present method of determining whether the electrical characteristics of a fabric are suitable for its intended use is to make the necessary electrical tests. This procedure for the development of coatings and fabrics having the minimum required electrical characteristics is costly, time consuming, and requires sensitive electronic equipment.

To facilitate the development of fabrics with the desired electrical characteristics, a correlation between the electrical and physical properties of fabrics was sought. The U. S. Army Natick Laboratories sent to the Bell Telephone Laboratories 17 fabrics to test for electrical properties. The following correlation between bulk density of the fabric and its dielectric constant was found (Fig 19).

The 17 specimens tested included coated and uncoated fabrics. The fabric surfaces represented varied from that of a smooth, glossy vinyl film, a matt surface chloroprene film, to that of napped, acrylic, uncoated fabric. Fiber types included were polyamides, polyesters, modacrylics, vinyls, and cellulosic. Some of the uncoated fabrics were made from yarns delustered with titanium dioxide. The coating compounds used on the coated fabrics included silicone, vinyl, chloroprene, and chlorosulphonated polyethylene. It can be seen that the relationship found in Fig 19 can be represented by essentially a straight line.

This correlation indicates that, given the thickness and weight of the fabric, the dielectric constant of the material can be estimated. The weight and thickness of a fabric can be determined from design information or can be measured on a finished fabric.

The information and correlation found in this study are limited to the specimens tested because of the small number of samples evaluated, and the lack of information on the ingredients used in the fabric coating compounds. According to Bell Telephone Company and Wheeler Laboratories, the relationship shown in Fig 19, while true for the materials tested, is not applicable to the high dielectric constant materials such as water and titanium dioxide. The latter is a pigment frequently used as a delustering agent in synthetic fibers. It should be noted that the correlation holds for cellulosic fibers having 6% hygroscopic moisture and for two acrylic samples containing small amounts of titanium dioxide as a delustering agent. This suggests areas in which additional work is required to fully understand the effect of radar transmission on composite structures combining high dielectric constant material with porous fibrous structures.

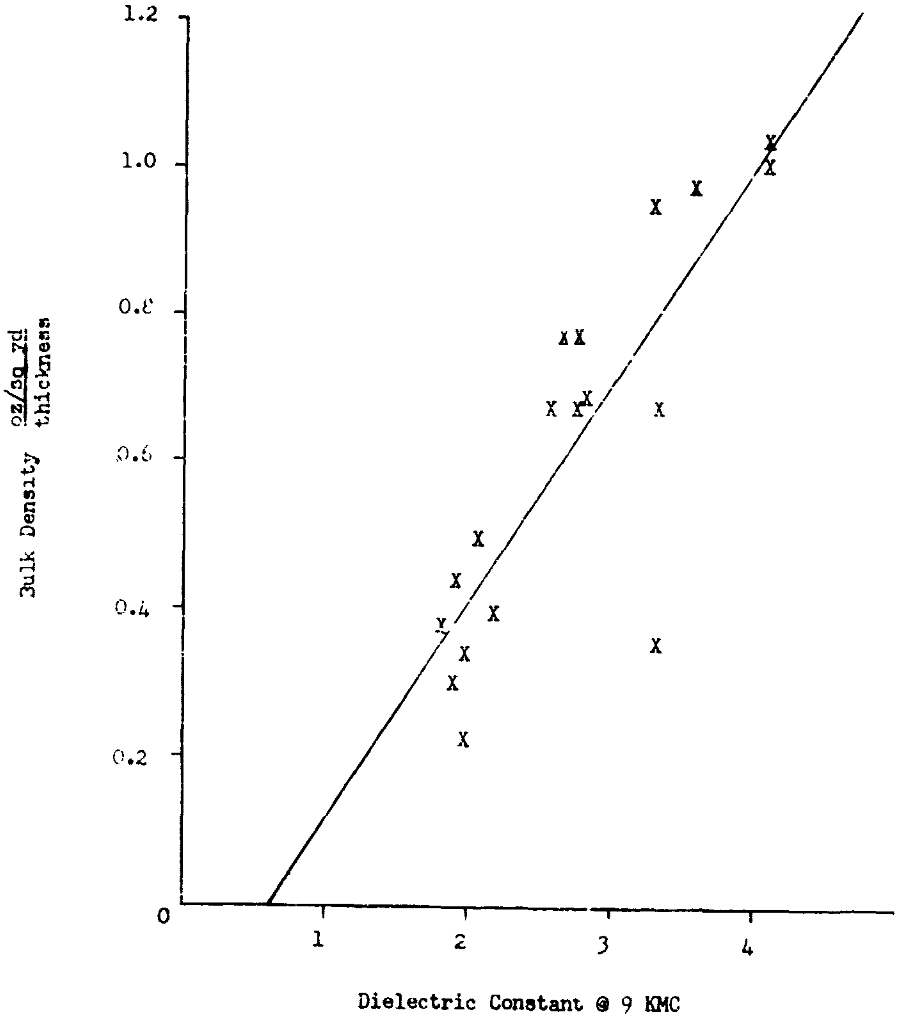

Fig 19 - Correlation Bulk Density and Dielectric Constant

Limited studies on the effect of radar transmission characteristics of carbon black and a number of metallic oxides frequently used in compounding chloroprene used in coating fabrics were reported by Cornell Aeronautical Laboratories (4). In general, it has been found that in order to obtain the best radar transmission characteristics, carbon black and metallic oxides used in the coating compounds should be well dispersed. This study apparently is also limited and further investigation in this area is warranted to determine which of the oxides and other chemicals used in compounding the coatings for fabrics have the least effect on radar transmissions through radome materials.

(5) Strength of Fabrics - The Army uses breaking strength tests for quality control of the fabric. Method 5100 of CCC-T-191, grab breaking strength, is used for control of the base fabric; and Method 5102, strip tensile, is used for control of the coated fabric. In these tests, the load is applied and measured in one direction and, depending on the direction of the applied stress, the breaking load in warp and filling is determined. As pointed out in the studies conducted by the Cornell Aeronautical Laboratory Report (4), the test methods for breaking strength used for the control of fabrics does not simulate actual use conditions. In actual use the air-supported tent is subjected to varying loading conditions--from constant uniform load due to internal pressure to varying but transient aerodynamic loads which may be at any angle to the direction of the fabric. The application of loads from aerodynamic forces can vary from a slow, gradual build-up to that of a sudden impact. Also, any vibration or motion of the tent fabric leads to motions approaching those of a whipping action. Hence, the strength of the material under various load conditions is important in the selection of the fabric for a given tent design.

A discussion of all aspects of the strength of fabric and other desirable mechanical characteristics is beyond the scope of this paper and much can be obtained from literature (1, 6, 7, 11, 12, 13, 14, 16, and 17).

It is the intent here to discuss some limited studies on biaxial stress of fabrics conducted at the U. S. Army Natick Laboratories. The work consisted of testing the burst strength of a lightweight fabric in the form of both a diaphram and cylinder. The diaphram burst test represents the loading condition of a fabric in the form of a spherical radome and the cylinder represents the loading conditions found in a cylindrical tent. Uniaxial breaking strength tests were also made as a point of reference. The stress data obtained for each condition of test are shown in Fig 20 and 21. Figure 20 shows the warp data while 21 shows the filling data.

In Fig 20, the uniaxial and 1-1 warp-to-filling load ratio, diaphram burst, show the highest strength and stiffness. In the cylinder burst test, where the warp was in the hoop direction (2-1 warp-to-filling load ratio), the fabric burst at a somewhat lower strength than either the 1-1 load ratio or the uniaxial test. In the cylinder test, where the filling

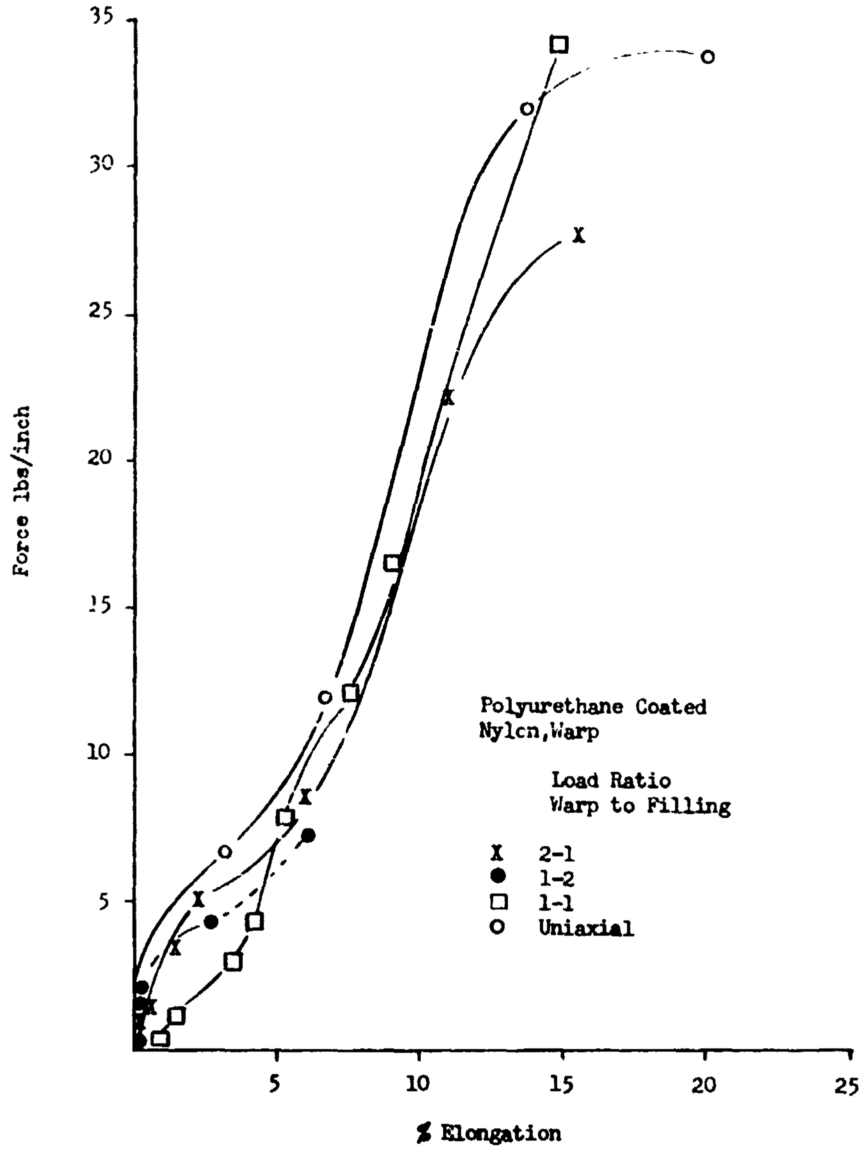

Fig 20 - Burst Strength, Warp

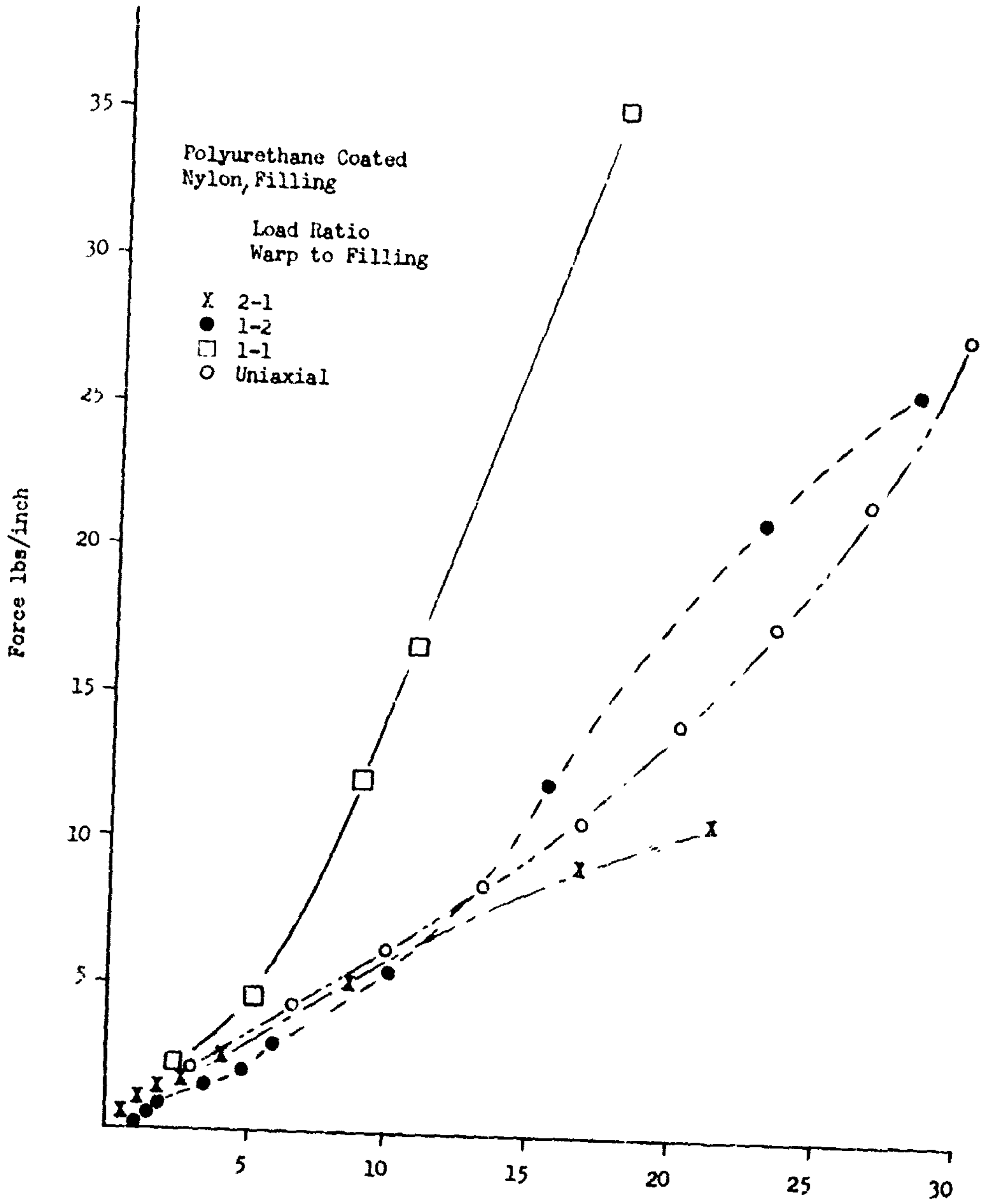

Fig 21 - Burst Strength, Filling

was in the hoop direction (1-2 warp-to-filling load ratio), the filling ruptured long before the full strength of the warp could be realized. The stiffness of the fabric appeared to remain the same for all modes of test. In Fig 21, the stress-strain behavior of the filling in biaxial stress showed the following: For the 1-1 warp-to-filling load ratio, the filling showed the highest strength and greater stiffness than the other methods of test. In the cylinder test, with the filling in the hoop direction (1-2 warp-to-filling load ratio), the strength and stiffness of the filling appears to be the same as the uniaxial test results. Where the warp was in the hoop direction (2-1 warp-to-filling load ratio), the warp ruptured before the full strength of the filling could be utilized. The above data suggest that, for load ratio of 1-1, radome construction a "square" fabric; i.e., a fabric with equal strength crimp and elongation in both warp and filling directions should be used. Studies conducted at Cornell Aerodynamic Laboratories (4) suggest that peak aerodynamic loads on spherical radomes can increase the constant uniform load, in the direction of greatest stress, by a factor of 2.1. However, since the aerodynamic loading can approach the fabric from any angle, one must make the fabric uniformly strong. Therefore, the premise for using a fabric equal in strength, crimp and elongation, in both warp and filling direction, is still valid. By increasing the uniform load due to internal pressure by a factor of 2.1, peak aerodynamic loads in spherical radomes would be accounted for.

From the geometry of an inflated cylinder and above tests, the stress due to internal pressure alone is twice as great in the hoop direction than in the longitudinal direction. This would indicate that the fabric for cylindrical tents should be designed with 1/2 the strength in the longitudinal direction. As indicated by the biaxial test data (Fig 20 and 21), the yarns in the longitudinal direction barely approached 1/2 their potential strength before rupture in the other yarn system caused the cylinder to fail. Thus, for stress due to internal pressure alone, excess strength in the longitudinal direction is not needed in cylindrical tents and adds to the cost and weight of the item without increasing its performance. However, to present the full condition of loading in a cylindrical tent, the aerodynamic loading must also be considered. Data on the aerodynamic loading of fabrics in cylindrical tents are currently being investigated (2, 20).

The above is a highly simplified version of matching fabric strength properties to the requirements of the tent. The obstacles encountered in determining fabric stress under dynamic conditions are well recognized, as are those for the development and production of fabrics to meet exacting end use requirements. These two objectives represent desired goals to be attained, rather than firmly established practices in common use today. The measurement of fabric stress under the dynamic conditions in the wind tunnel involves the development of techniques whereby stress concentrations in fabrics can be located and measured by direct or indirect means. The development of the scale for the aeroelastic parameter for fabrics in test models is the first step in this direction.

The attainment of the desired properties in the fabric is another subject which should be mentioned before concluding this paper. If fabric development were to keep pace with the development of air-supported tents, one can no longer afford the time required for this development based on traditional craft- - d experience. With few exceptions in the area of specialty fabrics, experience in producing fabrics to meet the requirements for air-supported tents still has to be attained. Even though the best fabrics available have been used in the fabrication of the tents developed for the Army, they were adaptations of good commercial fabric of multi-purpose use. The traditional method for selecting fabrics for air-supported tents was on the basis that they were rugged and strong enough to exceed the maximum calculated design load for the tent. Less attention was given to strength and crimp balance in the fabric whe. used for air-supported tents. This traditional approach to the development of fabrics for air-supported tents produces, at best, a compromise in fabric selection. The full potential of lightness in weight, durability, reliability, low cost, and size of the air-supported tent cannot be realized in this manner. It is suggested that fabrics for air-supported tents be designed utilizing information developed on the theoretical behavior of woven fabric which is subjected to biaxial stress. Some information in this area is available from literature (7, 13, 14, and 17).

This approach would provide the means for an engineering analysis of material for either predicting the performance of a given fabric, or for determining which fabric would be most suitable for a particular application or for the basic design of a fabric to fit a particular application.

Fabric engineering can only be as effective as the information relative to the desired characteristics of a fabric is known. Through wind tunnel studies, Military Characteristics, and other available information, the Army is developing the required characteristics for fabrics to be used for air-supported tents. The studies on fabric structure and stress analysis will be extended to the design and production of fabrics to meet the exacting mechanical, electrical, and other requirements for air-supported tentage.

In conclusion, therefore, the Army has shown eleven air-supported tents developed to meet specific operational requirements. The fabrics used by the Army were described and the following problem areas were discussed:

(1) Fabric coating compounds with improved flexibility at low temperatures.

(2) High-tenacity fibers with improved weathering characteristics.

(3) Studies on the dielectric constants on composite structures combining high dielectric constant materials with fibrous structures.

(4) Studies on the theoretical behavior of woven fabric subjected to biaxial stress conditions.

It is in these areas where additional research work is required to develop the full potential use for air-supported tents under all climatic conditions. Also, it was pointed out that the developments of air-supported tents and fabrics go hand in hand. It is felt that only through this approach can the full potential for lightness in weight and reliability of the air-supported tents be developed.

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MECHANICALLY MIXED POLYURETHANE FOAM-RIGIDIZED SOLAR COLLECTORS

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INTRODUCTION

The polyurethane foam industry has experienced spectacular growth in this country since World War II when the German patents and literature became available. Although the German data furnished the starting basis, the greater share of growth can be attributed to the original research of U.S. raw material suppliers, the consumer industry development of new products and methods, and government-sponsored research programs.

Goodyear Aerospace Corporation engaged in this early development work with polyurethane foams in 1946 on Air Force contract W33-038-ac15228. Subsequently, the polyurethane foam manufacturing technology has been perfected and applied to innumerable uses by the aerospace industry.

Approximately five years ago the company became interested in the potential use of foam for space application in an inflatable, rigidizable solar concentrator. Contract No. AF-33(657)-10165 was initiated last year to develop the foaming technology for such an application. This discussion will be essentially a status report of the effort conducted under this contract to date.

GENERAL FEATURES OF INFLATABLE RIGIDIZED SOLAR CONCENTRATOR

The inflatable-rigidizable concept for solar concentrator construction discussed herein is equally adaptable to a number of rigidization mediums, such as epoxy resins, gelatins, polyurethane foams, and others. This discussion is limited to a mechanically mixed polyurethane foam rigidization system.

The configuration shown in Figure 1 is applicable to a general space mission. During launch, the inflatable balloon will be packed in a container similar to that used for the Echo satellites. In space, the container will be jettisoned and the balloon inflated to a specific pressure. The concentrator portion of the balloon is a highly accurate paraboloidal shape that is tangent to a spherical end cap. A backflap is provided over the back of the concentrator portion to disperse and restrain the foam to the desired thickness and distribution.

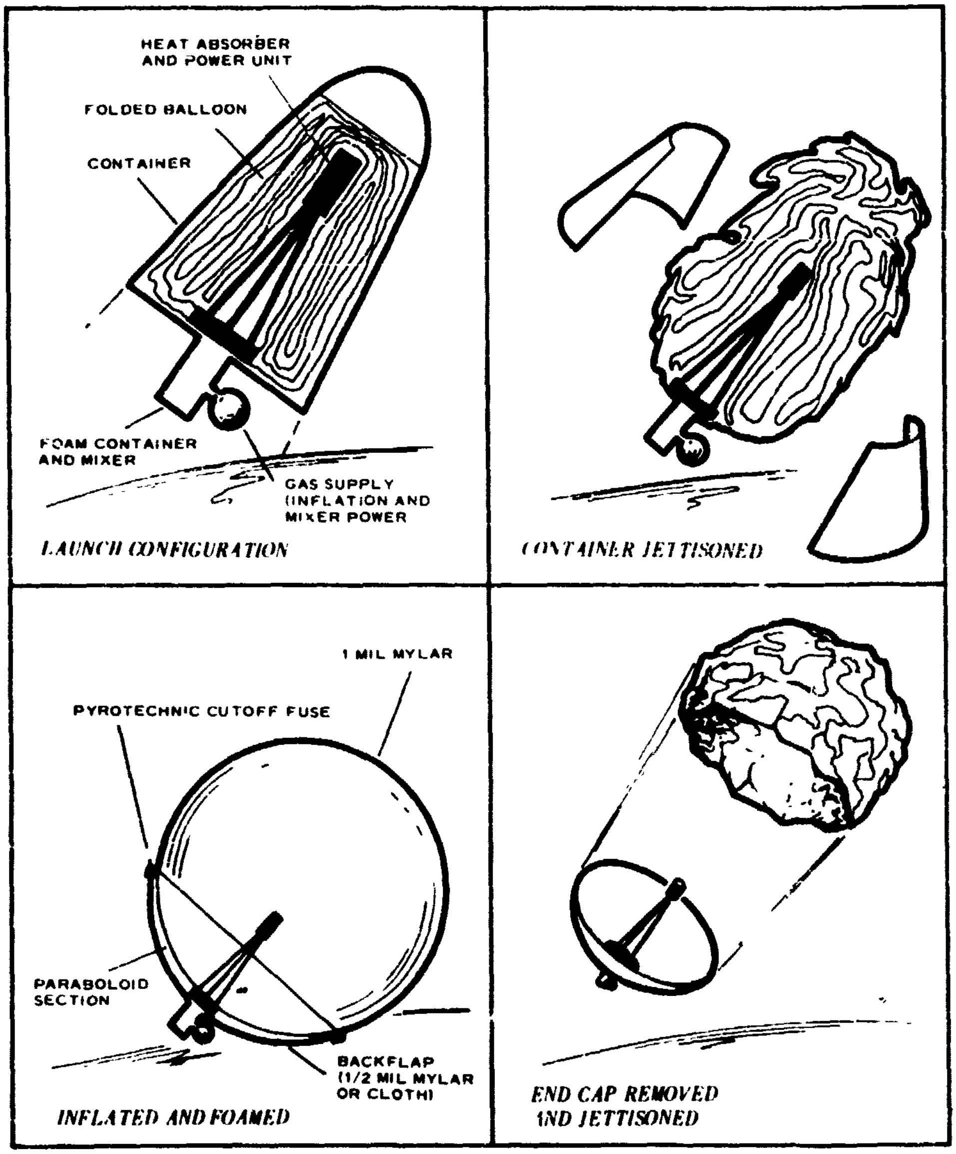

Figure 1 - Deployment and Rigidization of Solar Concentrator

The mirror portion of the balloon is constructed of one-mil Mylar with a vacuum-deposited aluminum reflective surface. The remainder of the balloon can be clear Mylar or coated depending on the passive thermal control desired during deployment. The backflap might be either 1/2-mil Mylar or lightweight nylon cloth, depending on the degree of gas venting desired for the particular foam. After the foam has been dispersed and rigidized, the end cap will be cut off by means of a pyrotechnic tape and jettisoned.

GENERAL FEATURES OF MECHANICAL-MIX FOAM SYSTEM

One of the mechanical mixer foaming systems currently in use is shown in Figure 2. A two-component system of resin and prepolymer in approximately equal proportions is used in this system. The resin is stored in two sealed Mylar bags between the mixer blades and the prepolymer occupies the remainder of the cavity. When the mixer is started, the resin bag is torn and shredded and the two components are thoroughly mixed. As chemical reaction occurs, the pressure rises in the container until the diaphragm bursts and the foam discharges into the annular opening in the hub that leads to the space between the backflap and the back side of the mirror surface film. As the foam continues to expand in the fluid state, it is forced out over the mirror surface film by the backflap restraint.

MAJOR PROBLEM AREAS

The problem areas reviewed herein are:

1. Effect of vacuum on foam during the chemical reaction
2. Achievement of foam distribution prior to rigidization
3. Control of exothermic reaction within acceptable limits
4. Determination of thermal and physical properties of vacuum foams to establish vital design data

EFFECT OF VACUUM ON FOAM

The major problem associated with foaming in a space environment is the effect of high vacuum. Polyurethane foaming by its basic chemical nature is accompanied by the formation of CO_2 gas during the cross linking of the polymer. Any gas in a high vacuum has practically an infinite expansion ratio. In

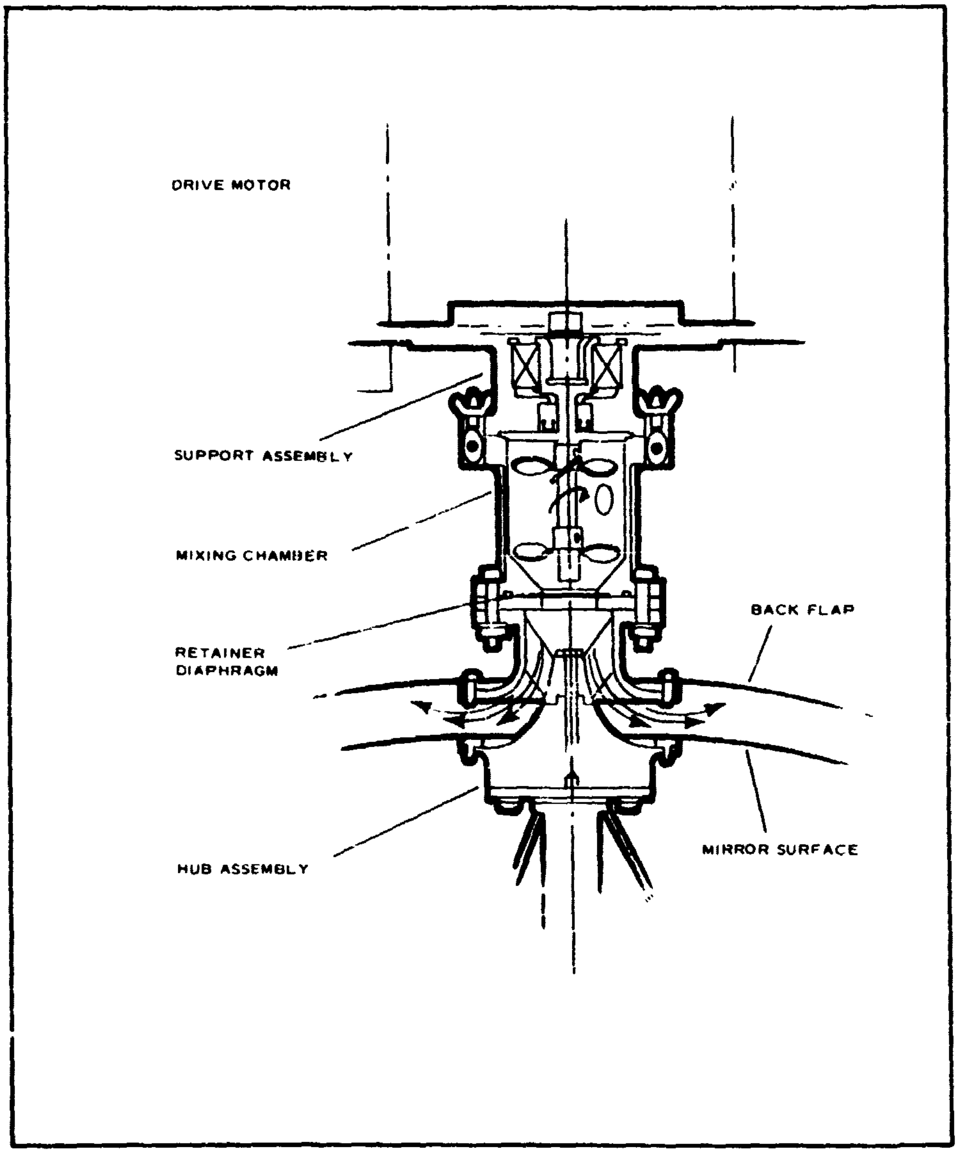

Figure 2 - Mechanical Foam Mixer

space, the only restraint to this expansion is obtained either from physical properties of the jelling fluid or from the container (such as the backflap over the mirror). These can be only rather limited effects that will be covered in the following discussion.

The polyurethane chemicals start out as a pair of viscous fluids that present very little resistance to the gas pressure forces involved. During the jelling process the foam cell size increases as the gas expands, more CO_2 is produced, and some cells coalesce when their intercell walls become thin and break through.

If rigidization occurs before the foam has collapsed from excessive bubble sizes, a foam is successfully achieved. The foam is an open-cell type if bursting of cells occurs just as the foam rigidizes. If the foam rigidizes without bursting the cells, a closed-cell foam is obtained. An excellent description of the mechanics of foam formation is given in Reference 1.

The open-cell foam usually is coarse and nonuniform in cell structure compared with the closed-cell type. This latter structure is more homogeneous and exhibits more adequate physical properties. So far it has only been possible to produce the closed-cell foam in a vacuum with the aid of some container restraint such as a backflap.

The drawback to the closed-cell structure that is produced in a vacuum is the difficulty in determining its physical properties in a normal atmosphere. Whereas the open-cell structure can be removed from vacuum for testing purposes without suffering collapse, the closed-cell structure at densities slightly below 2 pcf will be crushed by the pressure of one atmosphere.

The approach used has been to test the closed-cell foams at a range of higher densities, 2, 3, and 4 pcf, and to extrapolate the results into the lower density region. It is realized such an extrapolation might be questionable but a few limited tests have been made on open-cell, 0.6-pcf foams that were produced with less container restraint with excellent uniformity of cell structure. Unfortunately, it has not been possible to achieve consistently sufficient quantities for a comprehensive series of tests. The correlation achieved so far is shown in Figure 3.

FOAM DISTRIBUTION PROBLEMS

It was determined during tests in a vacuum chamber at 250,000-ft altitude, that there is a relatively short period (approximately, one minute maximum) in which rigidization should occur or the foam will collapse from excessive expansion of the gas. When the formula was changed to achieve a much briefer time to rigidization, the foam did not remain fluid long enough for proper dispersion over a surface. In cases of more violent gas evolution, a foam collapse again occurred from the very rapid expansion of gas. Some cases were observed also in which the fluid remained very liquid for an excessive interval and boiling rather than foaming took place.

Numerous attempts were made to reduce radically the amount of CO_2 gas

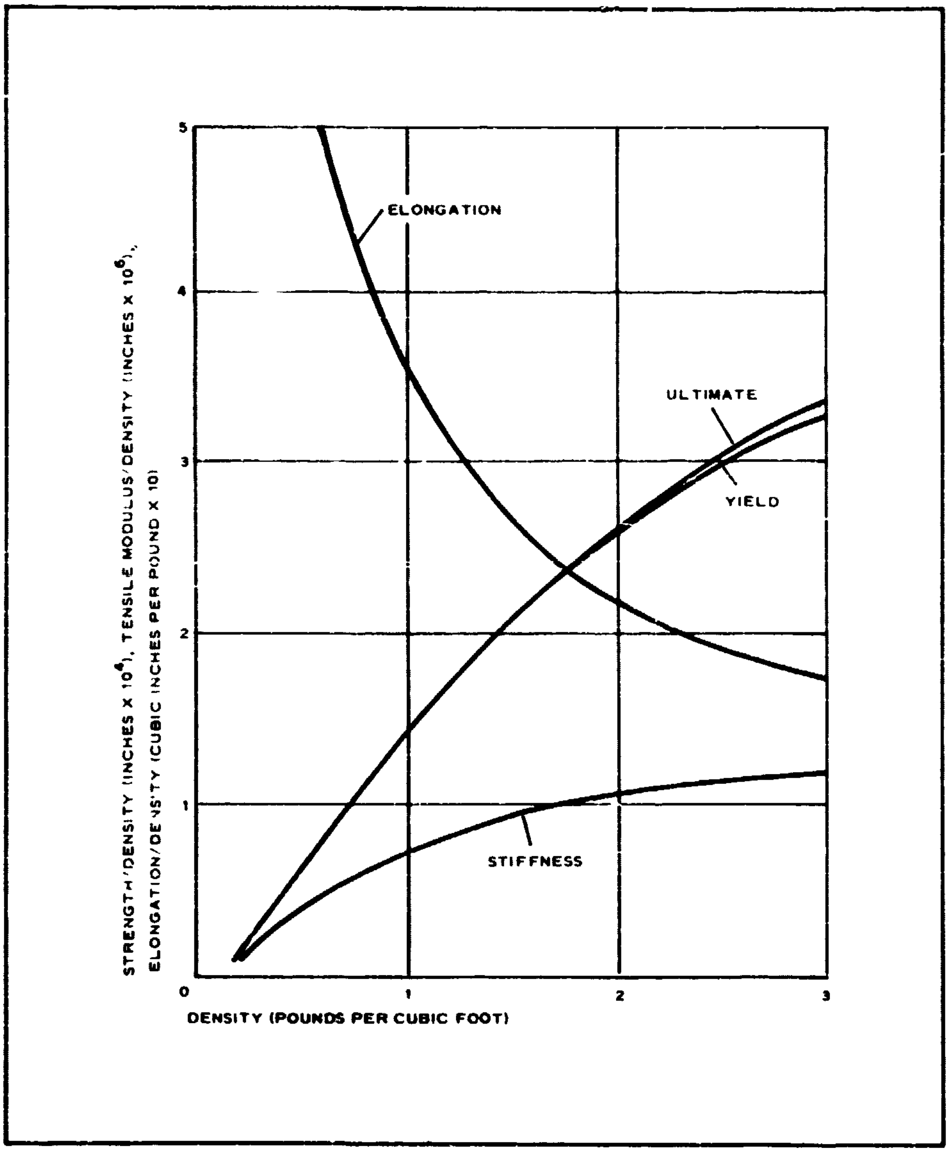

Figure 3 - Strength versus Density of Vacuum-Formed Polyurethane Foams

formation (to achieve a denser foam) by partial prereaction of the chemical components, by further distillation to remove as much residual moisture as possible, and by trying a variety of basic resins and prepolymers from several manufacturers. No observable improvement over the original formulation was achieved. With foam fluidity limited to such a brief interval, adequate dispersion of the foam becomes a problem.

The logical location for the foam generator is at the apex of the mirror. On early tests with a two-foot diameter mirror, which are covered in detail later, the dispersion was not satisfactory. The two-foot mirrors were made with a Mylar backflap set rather loosely. The foam rigidized at the rim and stopped further radial flow before foaming was complete, resulting in an excessively thick section of foam at the hub that did not extend out to the rim. When the backflap was pulled more taut, the expansion of the foam buckled the mirror surface inwardly because of the excess pressure and ruined its contour.

The next step consisted of using a tightly woven nylon cloth for a portion of the backflap based on the theory that this would relieve the excess back pressure. After a few tries this method was found to give consistent results.

A conceptual design of a mechanical means to achieve proper dispersion also was originated, as shown in Figure 4, but it was not evaluated after success was achieved with the nylon cloth backflap. By the mechanical concept, a ring collar would depress the backflap until it is fairly taut against the mirror envelope. Upon initial discharge of foam, the mirror contour might be depressed somewhat, but as soon as proper radial dispersion is achieved (while the foam is still in a fluid state), the collar mechanism could be unlocked and the backflap left free to expand to its full dimensions, thus permitting the inflation pressure to restore the mirror surface. This principle can be used to extend the maximum size limits of mechanical-mix mirrors, but additional work must be done to establish these limits.

Another scheme to achieve dispersion on larger size mirrors is shown in Figure 5. This method consists of directing the foam through Mylar ducting to remote areas of the collector. Thus dispersion is accomplished while the foam ingredients are still in a highly fluid and dense state. By proper sizing of the ducts and their discharge holes the proper distribution of mass quantities can be achieved. As expansion occurs during the chemical reaction, the foam patterns flow together and fill the backflap void. The ducting offers a secondary advantage by providing a harder core structural member that joins the hub to the lightweight foam.

Some limited small-scale testing of ducting systems has been conducted, using cylinders and panels to establish the feasibility of the concept. Additional work is planned to understand the design parameters involved.

DISCUSSION OF EXOTHERMIC PROBLEM

A typical exothermic temperature profile from current foaming tests with 2-ft-diameter models in a vacuum chamber is shown in Figure 6.

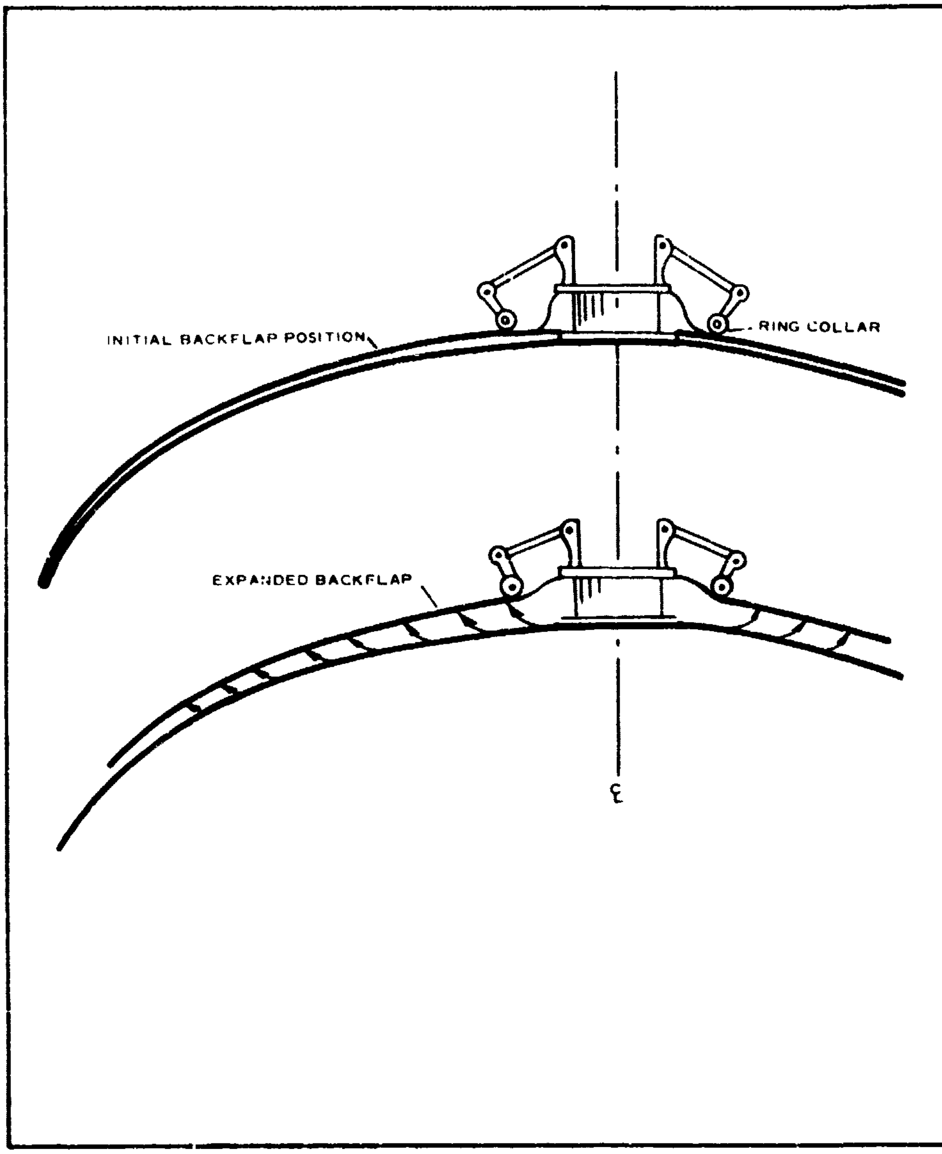

Figure 4 - Design Concept of Expandable Backflap

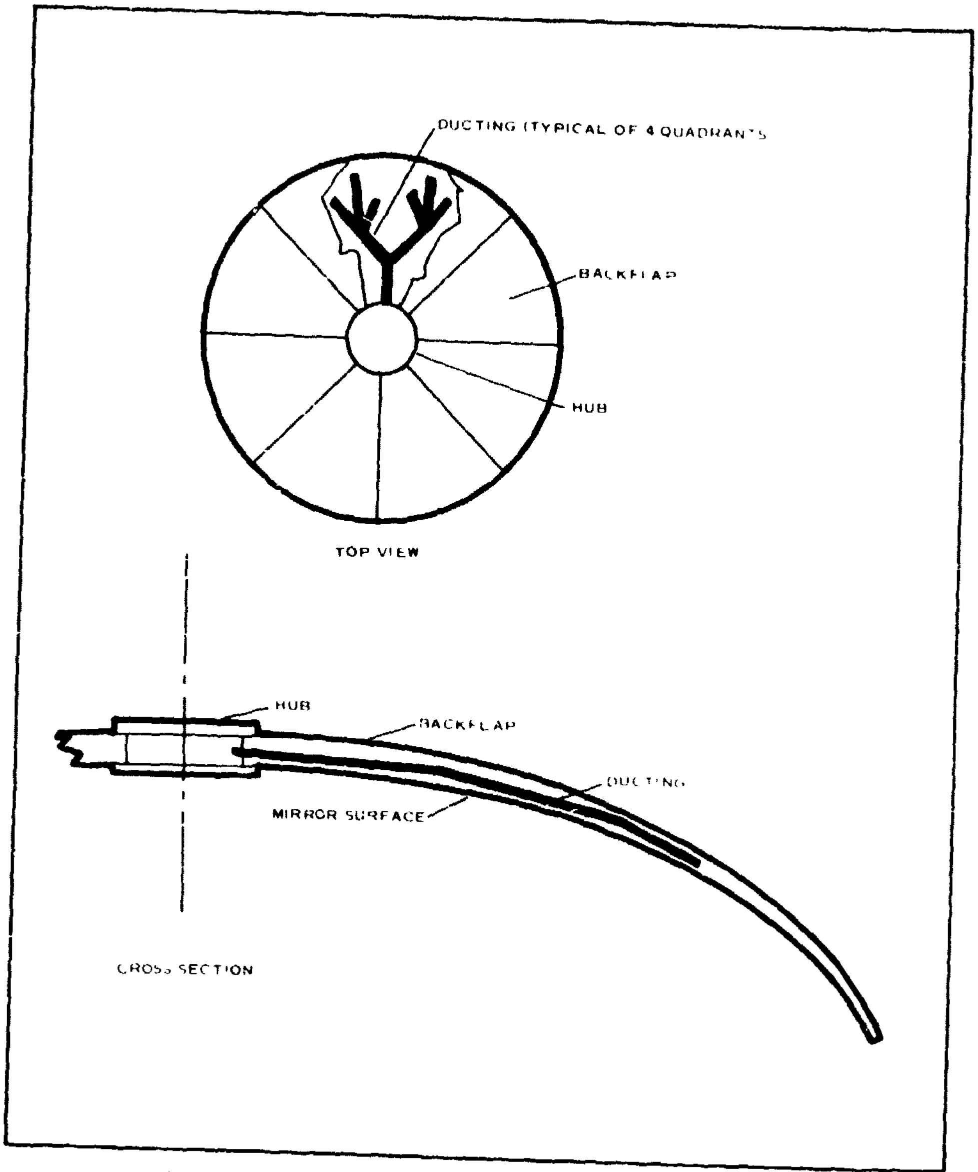

Figure 5.8 Ducting System for Foam Dispersion

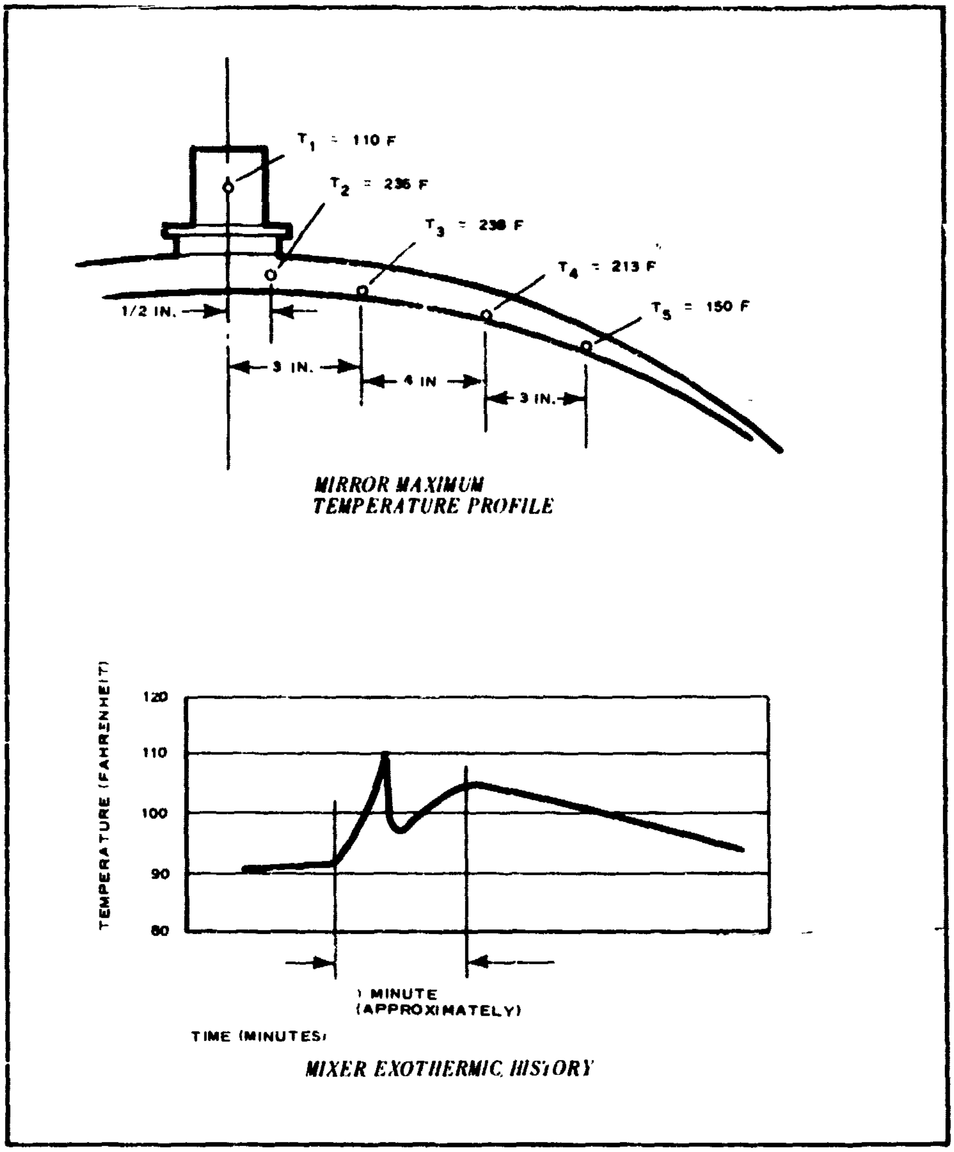

Figure 6 - Typical Foaming Temperatures

Note that maximum temperature occurs in the region near the hub. This can be attributed partially to the fact that all the foam originates at the hub and has to traverse this region to reach the rim, but it is mainly because the mirror is thickest in this region and the total heat produced is proportional to the mass of chemicals when all other factors are held constant.

It will be noticed also that some cooling effect occurs in the mixer immediately after diaphragm rupture. This might be due partially to the heat sink provided by the metal structure, but because of the rapidity of the temperature drop, it is believed more likely to be caused by the rapid expansion of the entrained gases.

The maximum temperatures that were experienced in these tests are within the current design pressure stress limits of the Mylar envelope, but a higher temperature seaming tape (Schjeldahl - GT-301) was found required after a few failures of seams in the early tests.

The Mylar film goes through a stretch and shrink cycle as the foam temperature increases and then decreases because of the decreased modulus of elasticity at higher temperatures.

Observation of the exothermic effects of variations in formula, indicates that only a limited temperature reduction can be obtained by such changes. The reaction time of foams produced under ambient pressure conditions can be slowed down by varying the formula but the same treatment of a foam produced in a vacuum results either in a collapsed foam or in one with poorer physical properties.

PROBLEMS ASSOCIATED WITH DETERMINATION OF FOAM PHYSICAL PROPERTIES

When attempting to make specimens for physical test at atmospheric pressure of foam produced in a vacuum, it was found that foam with a predominantly closed-cell structure and less than 2-pcf density would suffer radical shrinkage when removed from the vacuum chamber.

It was believed, therefore, that physical testing of such foam under ambient pressures would introduce a significant error in results that could not be corrected analytically with any degree of confidence. Hence, special test fixtures were designed to test the specimens in a vacuum.

Bulk foam quantities of approximately 2-, 3-, and 4.5-pcf density were then foamed in a vacuum chamber, removed just long enough to size and weigh the specimens accurately, and then again stored in vacuum until they could be tested.

Since foams actually produced in the two-foot-diameter mirror models were in the 0.5- to 0.75-pcf density range, it is desirable to extrapolate the physical properties data to this region. As mentioned previously the accuracy of this amount of extrapolation might be questionable. Fortunately, a

few bulk foams of 0.5-pcf density that did not shrink were produced during some quality assurance tests. These were predominantly open-cell type but of uniform cell structure approximating the closed-cell uniformity. These foams were tested and they furnished reliable data points for fairing in some extrapolations.

Unfortunately the open-cell foam usually produced by the current method has such large voids and nonuniformity that it is not suitable for test specimens.

PHYSICAL PROPERTIES TEST PROGRAM

To provide basic design data for a foam-rigidized solar concentrator, a number of physical properties of the foam must be established. The properties listed in Table I were considered the minimum required to establish reasonable confidence in a material. These properties are currently being determined in a test program

TABLE I - PHYSICAL PROPERTIES DATA REQUIRED

Structural	Optical-thermal-physical
Stress-strain	Surface reflectivity
Shear strength	Solar absorptivity-emissivity values of reflective surface
Foam to Mylar bond strength	Linear thermal coefficients of expansion
Creep strength	Thermal conductivity
Dimensional stability versus time	
Poisson's ratio	

STRUCTURAL PROPERTIES TESTS

Typical plots of tensile strengths and tensile modulus versus density at room temperature and at 185 F are shown in Figure 7.

Shear strength, bond strength, and Poisson's ratio tests have not yet been accomplished. A few creep tests have been initiated, but results are not available.

Dimensional stability versus time was conducted on one specimen. No significant distortion was observed but the data are too limited to warrant any conclusions.

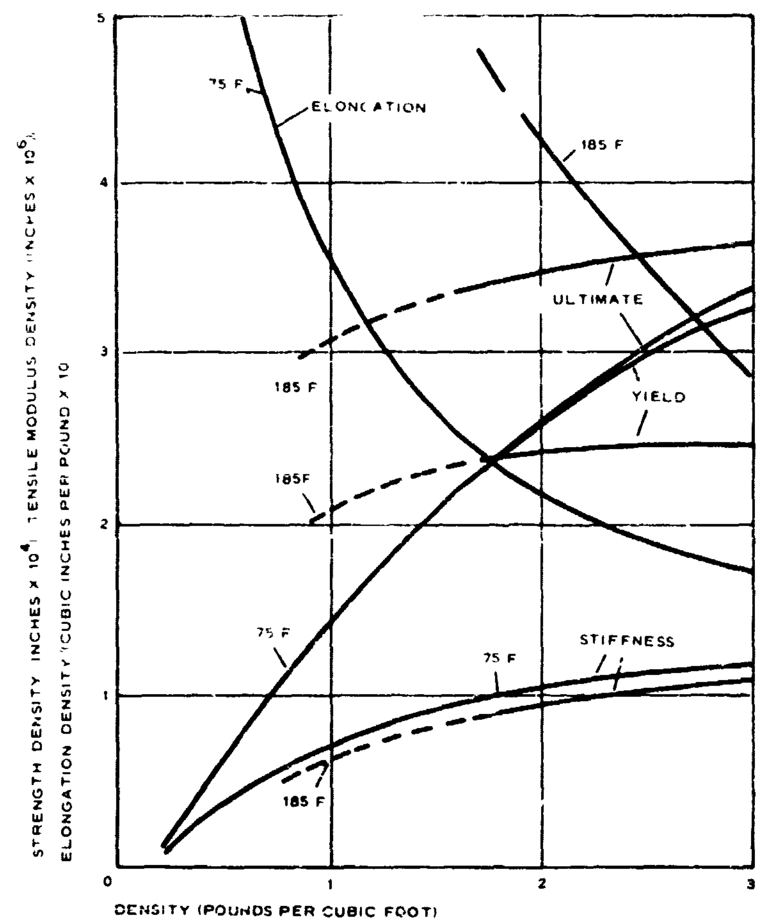

Figure 7 - Strength versus Density at 75 and 185 Fahrenheit

OPTICAL THERMAL PHYSICAL PROPERTIES

Reflectivity tests have been conducted with the Perkin-Elmer Model 98 monochromator in the wave length region from 0.3 to 2.5 . A curve of typical reflectivity achievable with production vacuum-deposited aluminum Mylar is shown in Figure 8. This figure also shows the reflectivity of a laboratory specimen of this same film coated by a proprietary Bausch and Lomb process to improve its emissivity. It is suspected that this coating is essentially silicon monoxide because the results are comparable with those reported in Reference 2. Additional work with films coated with varying thicknesses of silicon monoxide is currently being conducted with company research and development funds. The results of this work will be available shortly.

Mirror performance is concerned primarily with specular reflectance rather than total reflectance. Therefore, these curves are of interest primarily in determining efficiencies of a concentrator. Incidentally, foam-backed specimens of these same film materials exhibited practically the same reflectivity values. However, in thermal analyses of concentrator operating temperatures and thermal gradients, the total reflectivity determines how much of the solar heat flux is absorbed by this surface. (It has been found with the thickness of vacuum-deposited aluminum coating presently used, namely 600 to 2000 A, that transmissibility is a negligible quantity.) Since absorptivity is not amenable to any further significant reduction, it becomes evident that the greatest reduction of operating temperatures can be achieved by increasing emissivity values. This, of course, must be accomplished without seriously degrading reflectivity.

Determination of α/ϵ values was conducted on specimens suspended in the vacuum chamber of the solar simulator shown in Figure 9. The specimens used were reflective Mylar films applied to a thin disk of aluminum. The outside was exposed to a solar flux equivalent to two suns and the interior of the vacuum chamber was a nitrogen cold wall simulating outer space. With this setup a time-temperature-increase history was obtained on the specimens. The data obtained from these tests are shown in Figure 10. One curve represents the commercially available reflective film surface. The other curve was obtained from a special laboratory-prepared specimen with a proprietary coating. A major improvement results from the use of this coating. To arrive at absolute values for absorptivity and emissivity an evaluation method was used as discussed in detail in Reference 3.

Absorptivity values as low as 0.07 have been obtained, although 0.125 is a more representative value to be expected without premium expenditures to achieve the higher quality.

The laboratory specimen with the special emissivity coating exhibited an average absorptivity of 0.17, but it is suspected that the original film was very close to this same quality prior to processing and therefore did not degrade the material by even the 0.045 $\Delta\alpha_s$ indicated.

The emittance of the uncoated aluminized Mylar ranged from 0.045 to 0.065, whereas the specially coated films achieved 0.55 to 0.65. This drastic change in emissivity is the basic reason for the spectacular drop in equilibrium temperature of the latter test specimen, as indicated in Table II.

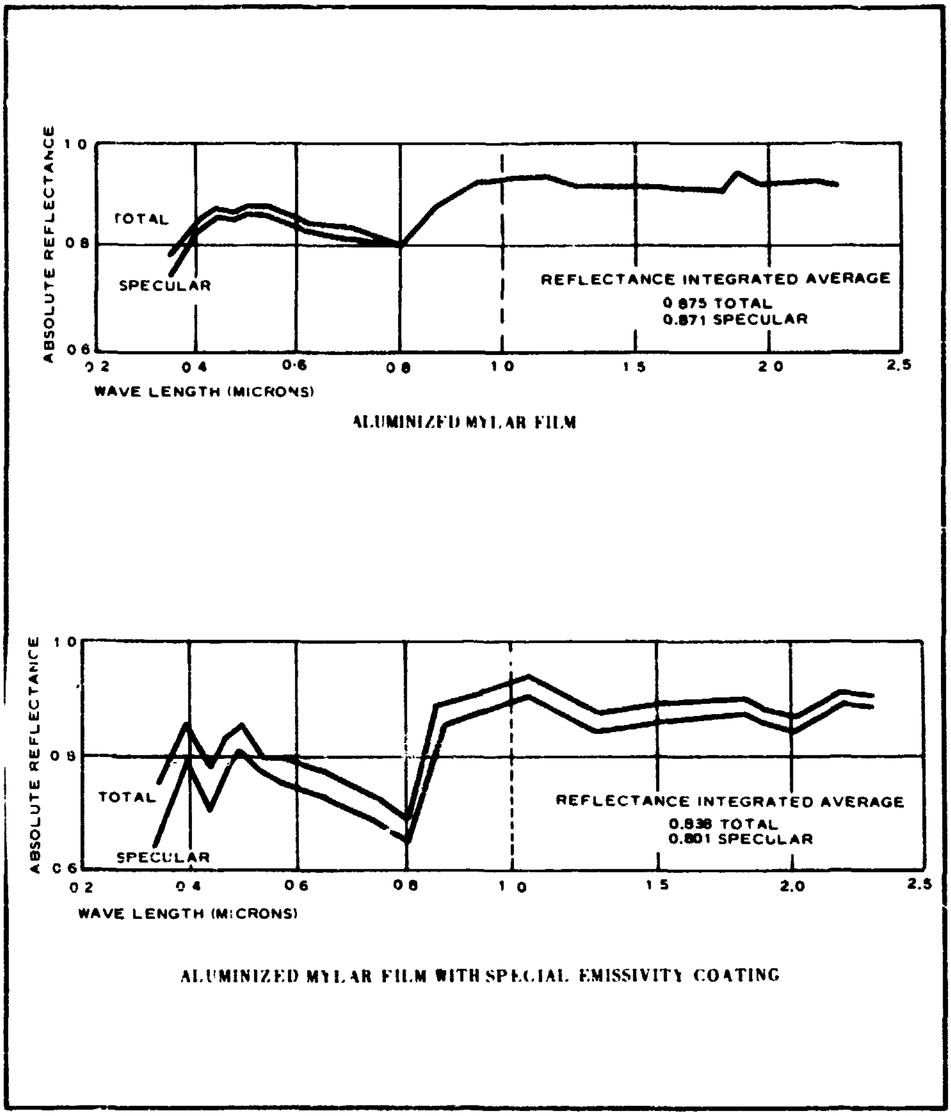

Figure 8 - Wave Length versus Absolute Reflectivity of Two Types of Reflective Coating

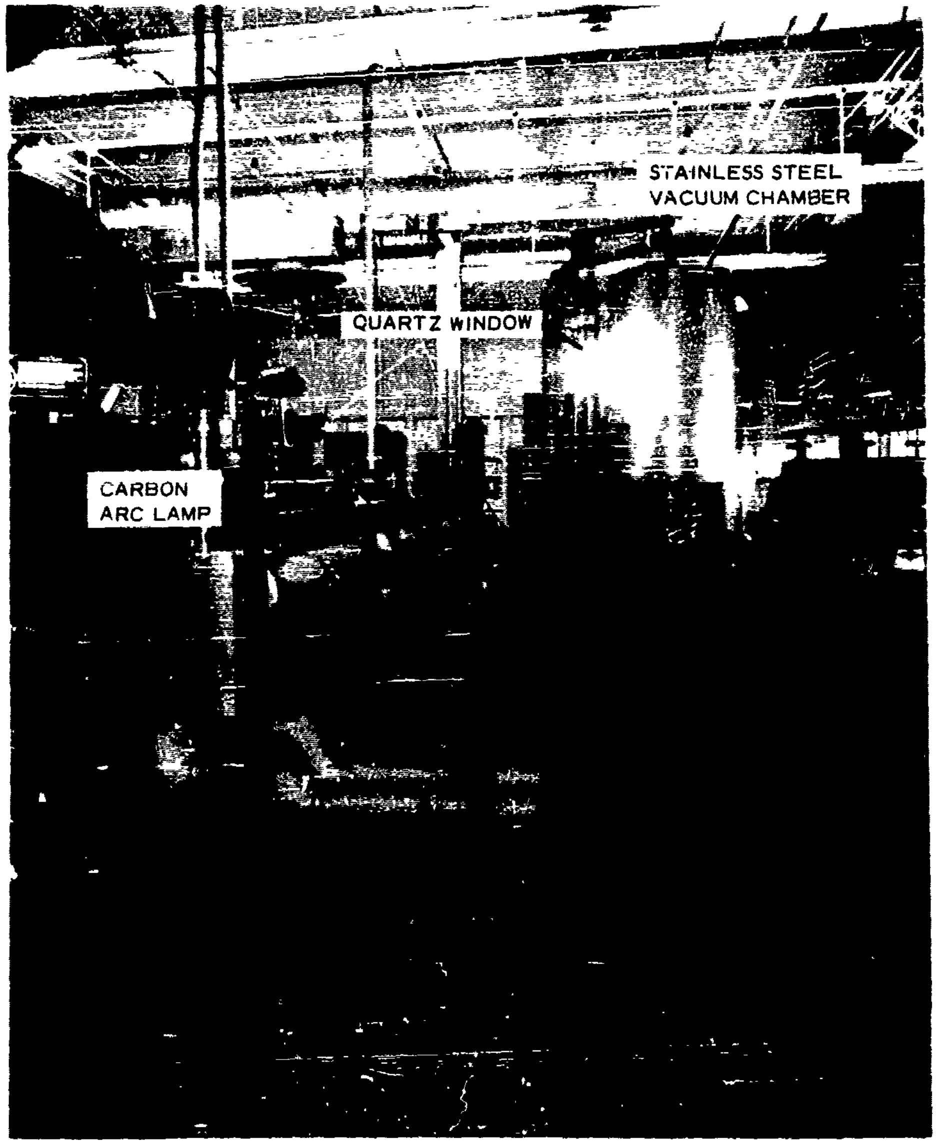

Figure 9 - Solar Simulator for Absorptivity and Emissivity Tests

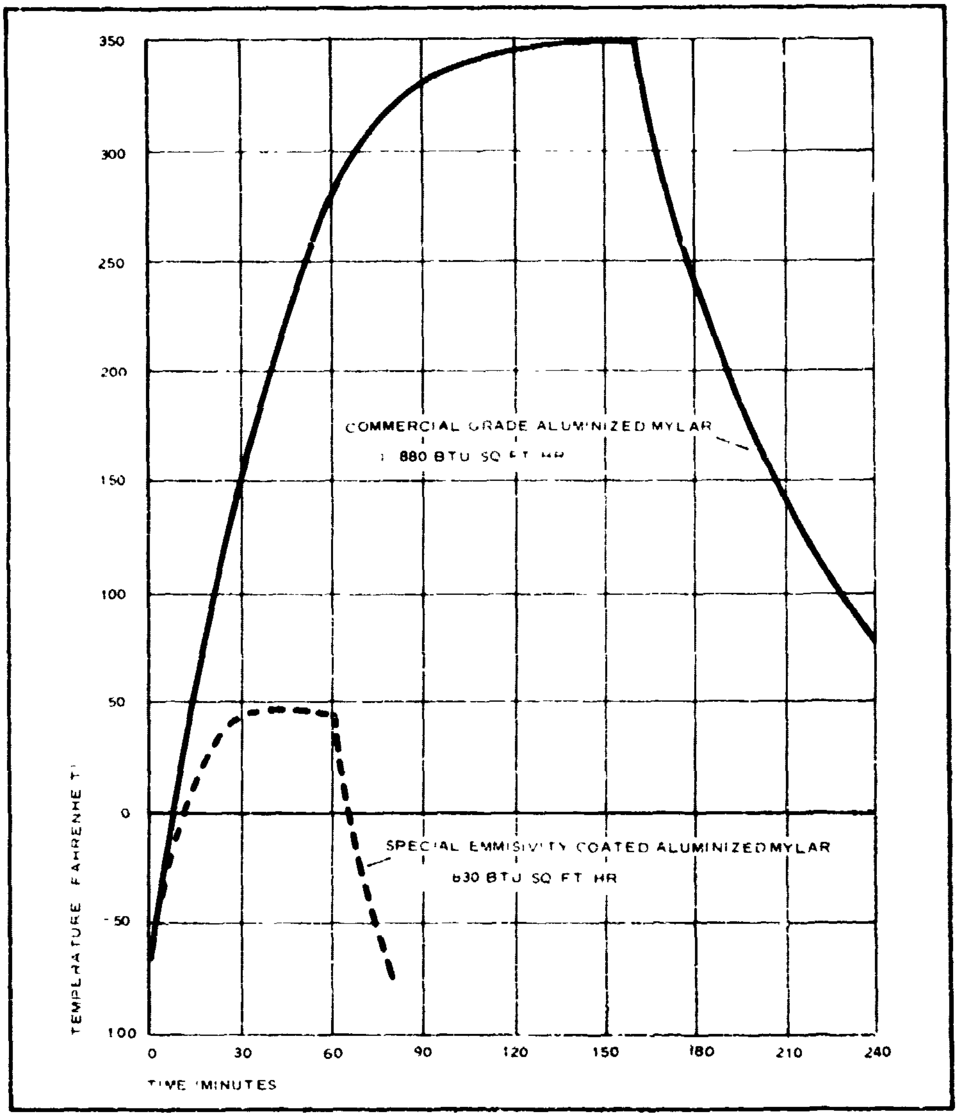

Figure 10 - Time-Temperature-Increase History of Two Types of Reflective Coating

Some typical orbital computations of front-face and back-face temperatures are indicated in Figure 11.

The case representing the specially coated surface shows peak temperatures and reasonable thermal gradients well within the limits of the foam and Mylar materials. The case representing commercially available aluminized Mylar is much higher in peak temperature and exhibits much larger thermal cycles. The data and analysis are not comprehensive enough to warrant eliminating the latter material, but they do indicate an extremely marginal condition.

TABLE II - SOLAR CONCENTRATOR RADIATION EQUILIBRIUM DATA

Material	Front face		Back face		Temperatures (F)				Conductivity/ thickness
					Maximum		Minimum		
	absorptivity	emissivity	absorptivity	emissivity	front face	back face	front face	back face	
	0.1000	0.0650	0.2000	0.8000	291.0	38.[illegible]	-161.5	-246.6	0.033
	0.1000	0.0650	0.2000	0.8000	274.0	42.8	-182.0	-243.2	0.050
	0.1000	0.0650	0.2000	0.8000	233.5	51.6	-206.5	-239.1	0.100
Uncoated aluminized Mylar	0.1000	0.0650	0.2000	0.8000	184.5	60.5	-221.0	-237.8	0.200
	0.0700	0.0450	0.2000	0.8000	277.0	37.8	-194.5	-260.3	0.033
	0.0700	0.0450	0.2000	0.8000	254.5	41.5	-212.5	-258.3	0.050
	0.0700	0.0450	0.2000	0.8000	205.5	48.8	-232.5	-256.2	0.100
	0.0700	0.0450	0.2000	0.8000	153.5	53.9	-243.5	-255.6	0.200
	0.1500	0.5500	0.2000	0.8000	51.0	22.5	-58.5	-195.1	0.050
	0.1500	0.5500	0.2000	0.8000	49.5	27.7	-71.5	-170.3	0.100
Aluminized, Mylar coated for emissivity improvement	0.1500	0.5500	0.2000	0.8000	47.0	29.8	-85.5	-149.7	0.200
	0.1500	0.5500	0.3000	0.3000	59.5	119.6	-51.0	-149.6	0.050
	0.1500	0.5500	0.3000	0.3000	63.5	115.1	-58.0	-125.1	0.100
	0.1500	0.5500	0.3000	0.3000	69.0	110.1	-64.5	-105.9	0.200

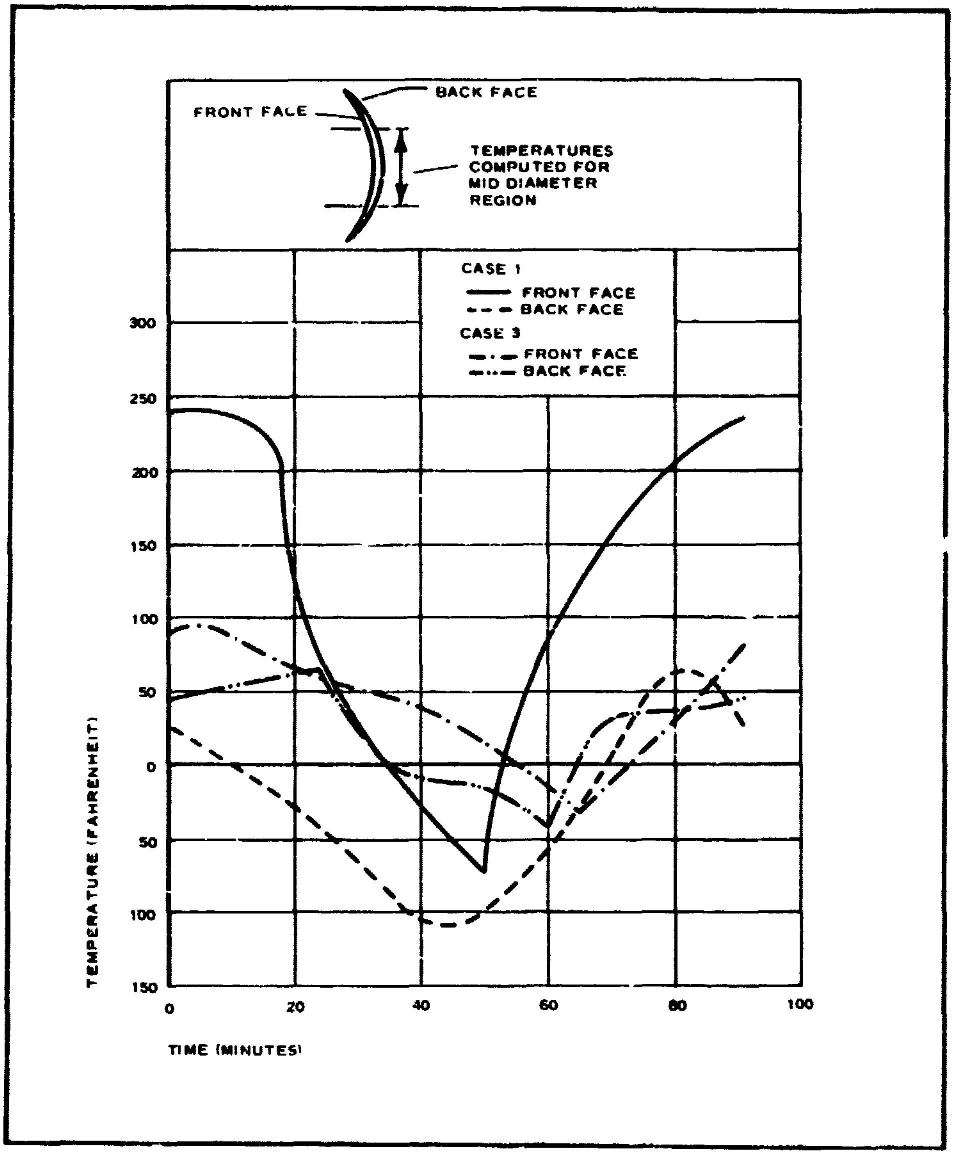

Figure 11 - Orbital Computations of Front- and Back-Face Temperature

THERMAL CONDUCTIVITY OF FOAM

Thermal conductivity tests of foams are now being conducted with a Wyle laboratory Model 140A analyzer, modified to place the guarded hot plate in a bell jar. Early tests of foams in the 2.0-pcf density region have demonstrated values of 0.04 for K. It was at first suspected that this value was too high for a completely evacuated foam, but specimens that were left in the bell jar for a period of time at 10^{-6} Torr and did not show any significant decrease.

HEAT DISTORTION TEMPERATURE

Quite early in the current program, it was found that the heat distortion point was a major criterion in the foam selection process. It was found also that the test to determine linear thermal coefficients of expansion could be used as a screening test to permit rapid evaluation of the distortion factor. The test setup is shown in Figure 12. In this test, the temperature of the specimen is raised gradually, its elongation is observed periodically with the cathetometer, and the data are plotted as elongation versus temperature. A curve of the type shown in Figure 13 is generated.

Foams that exhibit a flatter slope at the left end of the curve will be more susceptible to thermal gradient distortion and those with a lower knee to the curve will necessarily be restricted to lower operating temperatures.

It was discovered also that the general characteristic of the curve was quite independent of density in the region of interest and with the formulas that were investigated. The elongation versus temperature test therefore becomes a useful quality control tool for detecting inadvertent deviations in processing, errors in formulation, and deterioration of ingredients.

RIGIDIZATION OF TWO-FOOT-DIAMETER MIRROR UNITS

The test installation of a 2-ft mirror unit in a 250,000-ft altitude chamber is shown in Figure 14. A modified unit soon to be tested in RTD'S high vacuum chamber (10^{-6} Torr) is shown in Figure 15.

Some mirrors that were foamed in the Goodyear Aerospace vacuum chamber are shown in Figure 16. In these mirror rigidizations, the contour of the inflated envelope is checked prior to foaming, after the foam has rigidized, and again after the envelope inflation pressure has been released by cutoff by a device, as shown in Figure 17. The purpose of the several measurements is to determine the effect of foam rigidization and spring back on the original contour. With these data the envelope can then be previously contoured by proper offsets to achieve a final true paraboloid. A typical plot of the data obtained is shown in Figure 18.

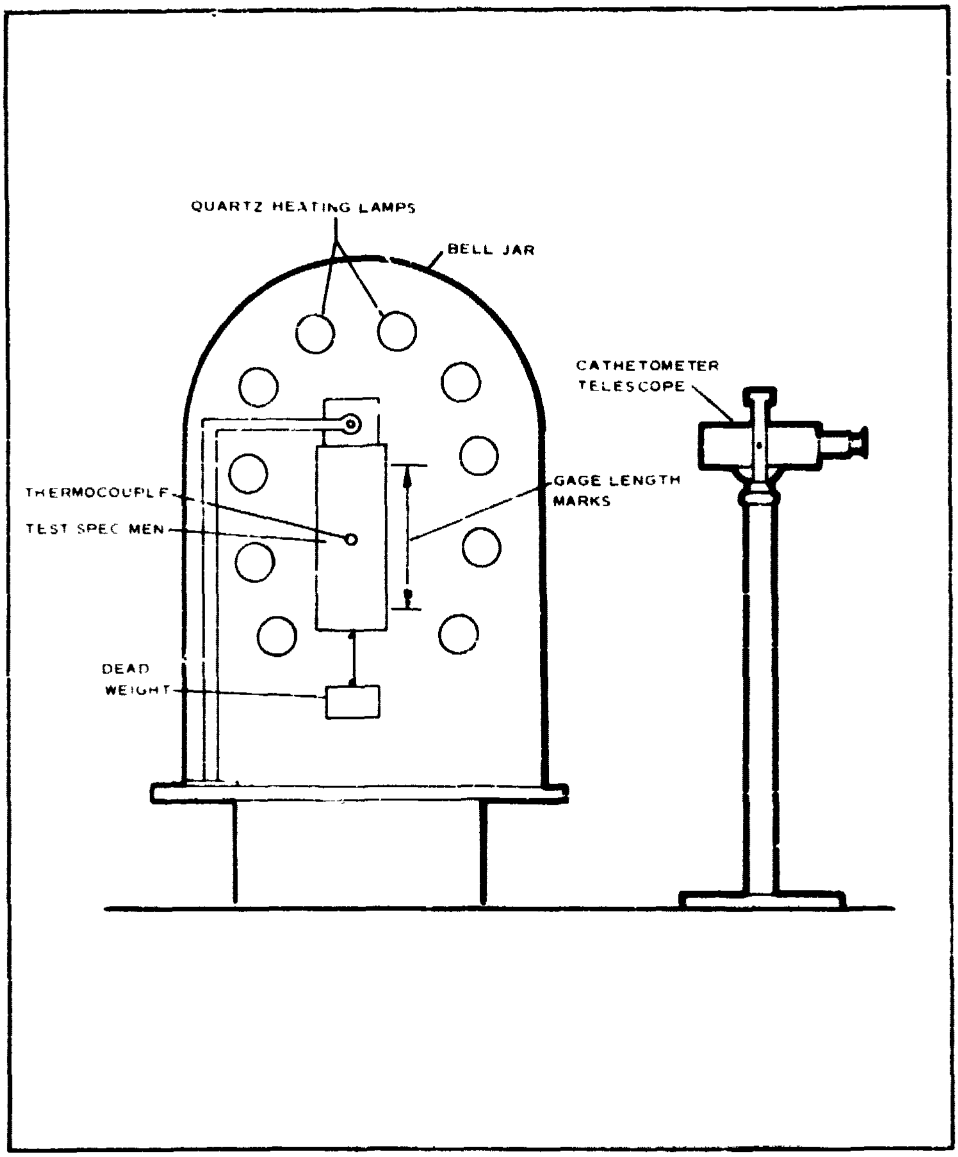

Figure 12 - Test Setup for Measuring Linear Thermal Expansion

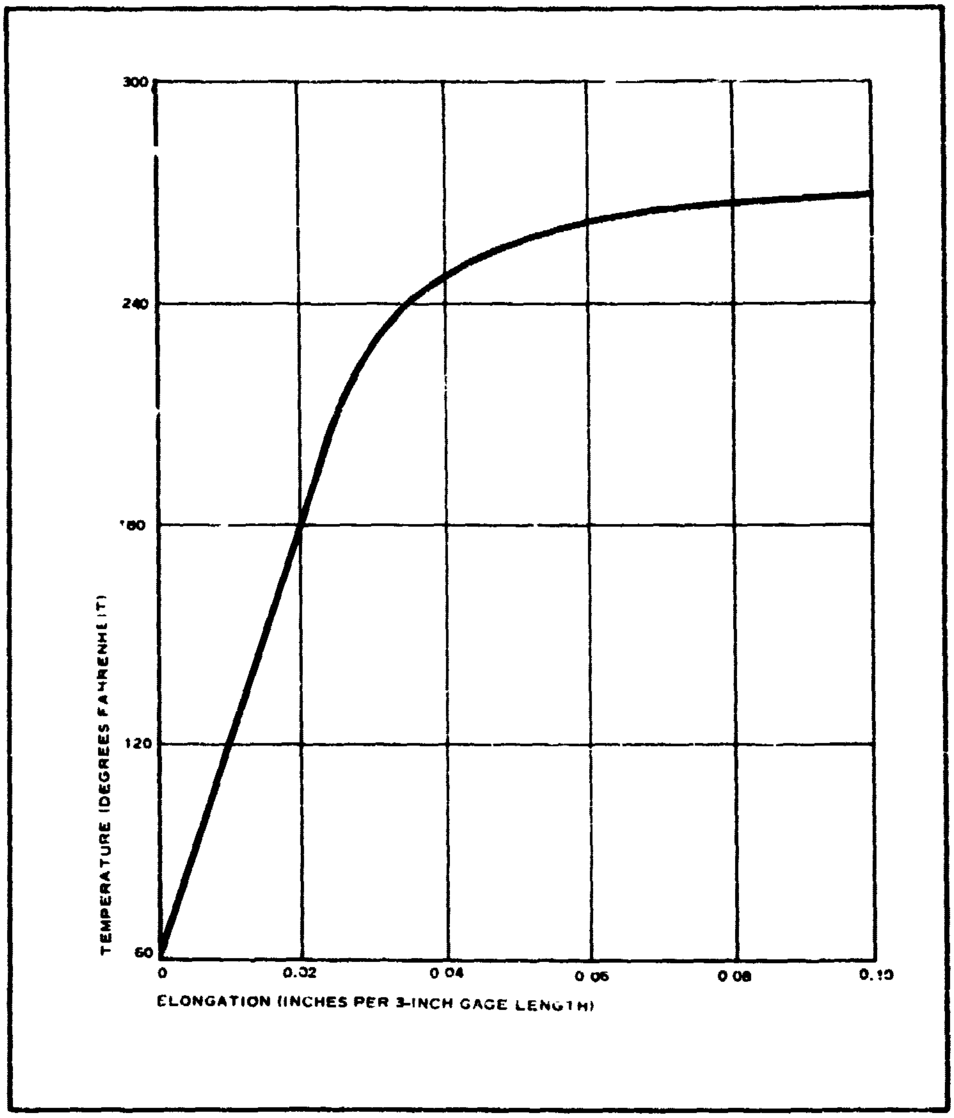

Figure 13 - Elongation versus Temperature Characteristic of Vacuum-Formed Polyurethane Foams

Figure 14 - Two-Foot Mirror in Arc Tunnel Vacuum Facility

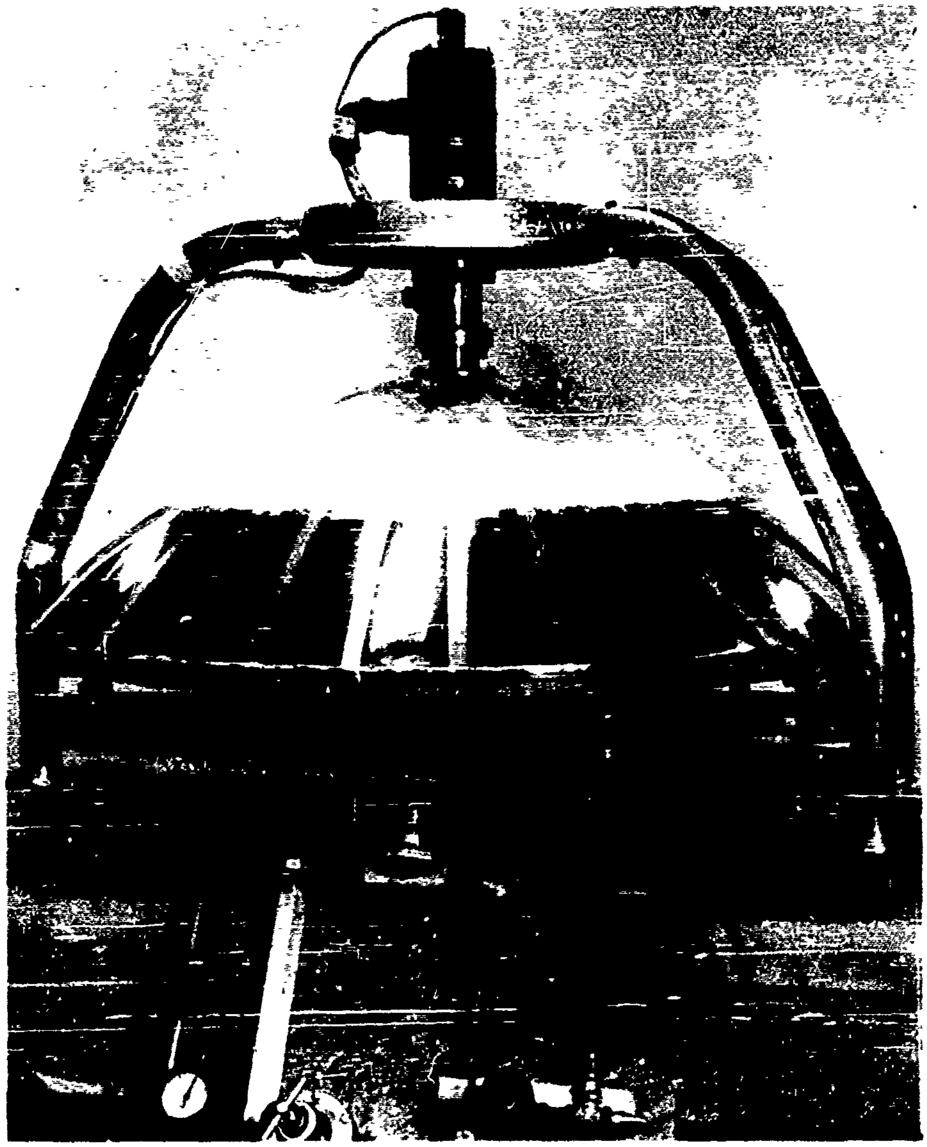

Figure 15 - Modified Unit to be Tested in RTD High-Vacuum Chamber

Figure 16 - Mirrors Foamed in Goodyear Aerospace Vacuum Chamber

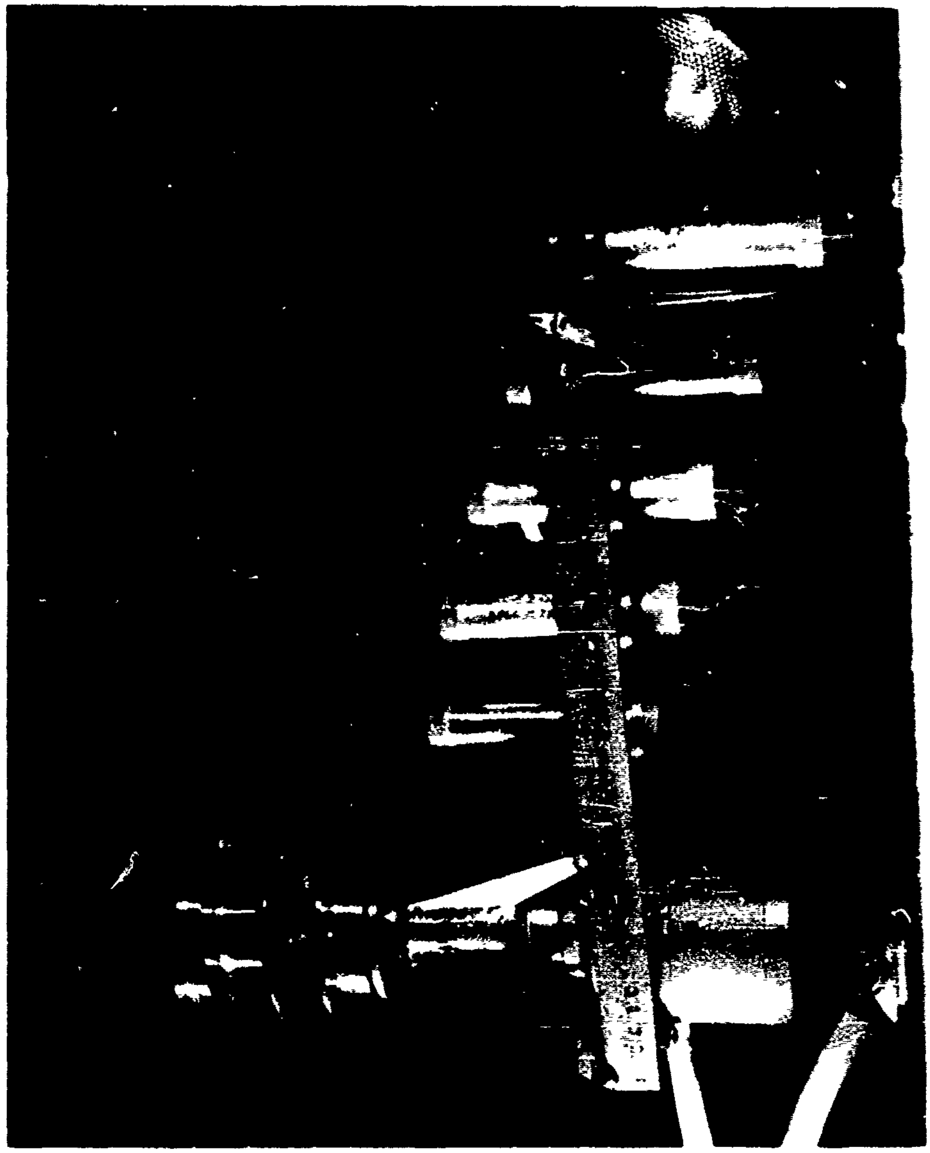

Figure 17 - Contour Checking Device

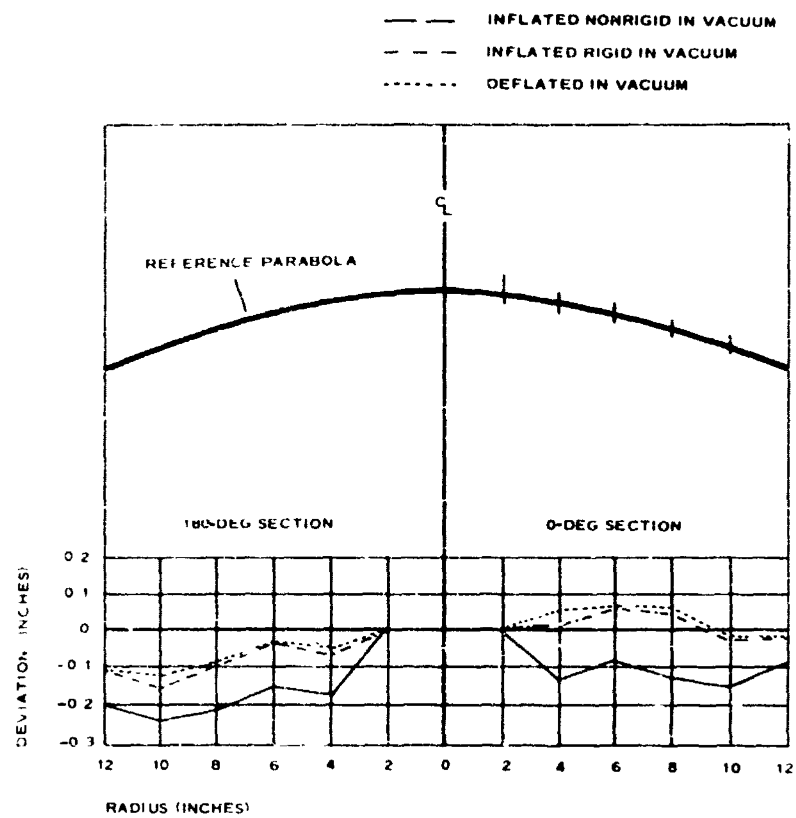

Figure 18 - Typical Deviation from True Paraboloid

In addition to the dimensional checks, the following items are monitored:

1. Mixer power and speed
2. Mixer foam pressure
3. Mixer foam temperature
4. Exothermic temperature profile in foam
5. Thickness of foam

Several mirrors, some of which are shown in Figure 16, have been made that suffered only very slight shrinkage when removed from the vacuum chamber. These were made with foams of predominantly open-cell structure. However, the mirrors that have exhibited the best reflective surface by visual observation while still in the vacuum were predominantly closed-cell foam types. These mirrors are crushed by the pressure of one atmosphere and are unsuitable for direct comparison after removal from the chamber.

CONCLUSIONS

Additional work is required in the development of inflatable rigidized solar concentrators in the following areas:

1. Investigate further the duct dispersion method to establish design criteria
2. Continue larger scale mirror evaluations in vacuum
3. Achieve solutions to the problems of testing a mirror while still in the vacuum chamber or achieve a foamed mirror that will not crush when removed
4. Investigate reflective films to improve reflectance, α/ϵ ratios, and physical properties
5. Improve seaming methods
6. Continue work with foam physical properties

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DEVELOPMENT OF A ONE PART "FOAM-IN-SPACE" POLYURETHANE

BY

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INTRODUCTION

In 1958 the Materials Engineering Branch, Applications Laboratory of the Aeronautical Systems Division of Wright-Patterson Air Force Base initiated a number of in-house exploratory investigations to determine the feasibility of producing polyurethane foam at reduced atmospheric pressures, simulating outer space conditions. These tests included mixing of conventional urethane foam components in a vacuum and the use of catalyst-water systems absorbed on silica gels. The results of this preliminary work indicated that polyurethane foams could be utilized at a reduced atmospheric pressure to produce such objects as rigidized inflatable structures, lunar shelters, furniture, thermal insulation, shock absorbing devices, etc. However, it was also demonstrated that conventional formulations and processing techniques would not be suitable for use under these conditions.

As a result of the preliminary work, which established feasibility of the vacuum foamed material, a formal request for proposals was issued by the Aeronautical Systems Division to a number of organizations known to have the desired capabilities. The specific requirements in the directive were as follows:

a. The material must be based on polyurethane chemistry.

b. It must foam reliably in a vacuum environment - with the material directly exposed to the vacuum, of approximately 160,000 feet.

c. The foamed material should be approximately a 2 lb/cu ft density.

d. If pre-mixed material was developed it should have a minimum shelf life of 2 months at room temperature, i.e., it could not be cryogenically inhibited.

e. It should also foam at normal ambient pressures.

f. A minimum of mechanical equipment should be used during the foaming process and no special equipment was to be developed.

g. Material capability would have to be demonstrated by fabrication of two types of foamed structures, made in a simulated space environment. One was to be an expandable structure, such as a 7 ft diameter balloon, inflated and rigidized with walls 1 to 2 inches

thick under conditions simulating an unmanned spacecraft. The other structure, used to demonstrate large mass capability, was to be a full sized chair, made under the same conditions but not necessarily in the same manner.

Hughes' Aircraft Approach

After the receipt of the contract the Hughes Aircraft Company, on analysis of the problem, came up with several approaches, differing widely bacause of the vast disparity between the two structures.

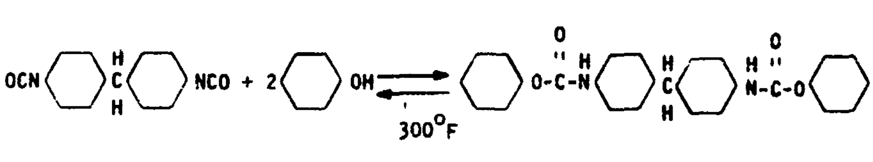
a. In order to foam rigidize the balloon a one part, pre-mixed and pre-distributed material was believed best, since this would require no special vacuum operated equipment to be developed.

b. On the other hand, the chair, because of its size and shape, (an approximately 2 x 2 x 2 ft block with a back and arm rests) was well adapted for fabrication by metering and mixing a two component material into a mold. Such a procedure, however, would probably have required special equipment in order to operate in a vacuum.

On consideration of all the factors involved in production of each type of material it was decided that if the one part material were produced it would be a very simple approach, and one which could be used for both applications, as well as being potentially a very useful material. It was therefore decided to concentrate solely on production of the one part, premixed foamant. It was also decided that heat would be utilized as the activator. Further, no mechanical equipment was to be used in the vacuum. The basic polyurethane reactions; i.e. diisocyanates reacting with a diol and/or a triol, and also reacting with water to produce CO_2 gas, would be adhered to.

Foamant Development

In the development of the desired one component compound the first step was believed to be a method of deactivating the isocyanate component. This was necessary since in common isocyanate-polyol mixtures the reaction takes place within 5 to 30 seconds, when catalyzed, and not much slower when uncatalyzed. This reactivity, of course stems from the isocyanate, so efforts were concentrated on this component. Two methods were tried. The first consisted of chemical blocking of the isocyanate with a material which would cleave off when heated, thus regenerating the original isocyanate. An example of such a compound, commercially known as Hylene MP, is shown below:



A number of similar compounds were prepared at Hughes using both toluene diisocyanate (TDI) and diphenyl methane diisocyanate (MDI) as the isocyanates and blocking agents such as acetyl acetone, diethyl malonate, phthalamide and others. A number of blocked compounds were successfully prepared.

However, in use it was found that the cleavage temperature in each case was too high (approximately 325 to 400°F), and when reacted in vacuum, the release of the blocking agent resulted in too voluminous a gas evolution. Also, complete stability was not obtained when the blocked isocyanate was mixed with the polyol and stored at room temperature.

The second method of deactivation tried was an attempt to encapsulate the isocyanate in a heat rupturing encapsulant. By this technique it was hoped that the encapsulated isocyanate could be mixed directly with the polyols with no reaction until heat was applied. Contacts were made with all the encapsulators in the United States and it was found that no one was prepared to encapsulate liquid isocyanates without a long research program. The National Cash Register Company, however, did agree to furnish solid MDI encapsulated in either of two thermoplastic materials melting at 122° and 140°F. In tests it was found that only the 140° material would resist damage during the mixing with the polyol. However, in attempting to make a foam using the encapsulated MDI and conventional polyols it was found impossible to get foams much lower than 10 to 15 lb. density and in addition the foams were of fairly low strength and poor quality. It was assumed this was due partly to the plasticizing effect of the encapsulant, which was present to the extent of 20% by weight of the MDI, and partly to the solid isocyanate which was used. The reaction was exothermic, however, and it is surmised that satisfactory results might have been achieved with other formulations and more extensive work. Therefore, because of the difficulties encountered in attempting to deactivate the isocyanates, this line of attack was discontinued, in favor of another technique, described below.

Concurrently with the work being done on the isocyanates a number of tests were also made on techniques for furnishing water to the system, since this was also considered to be a major problem in development of a one part mixture. This investigation of water sources was also carried out along two lines. The first consisted of investigating a number of hydrates, such as H_3BO_3, $MgO.H_2O$, $BaCl\ 2H_2O$, etc., which would release water of crystallization when heated. The second technique utilized Linde molecular sieves, which would also release their water when heated, or in a vacuum. The results of the first tests indicated that boric acid could be mixed with a liquid isocyanate prepolymer for at least a month and a half with no apparent action, when held at room temperature. When heated the isocyanate-water reaction took place very readily at temperatures of 200 to 300°F. The results secured with the molecular sieves were not so definite, however. The hydrated sieves appeared to store satisfactorily at room temperature when mixed with the isocyanate prepolymers. However, when the mixtures were heated in a vacuum considerable bubbling occurred, which apparently was due to the volatiles in the isocyanate prepolymers, rather than the formation of CO_2. Repeating the tests with devolatilized prepolymer showed no evidence of reaction, although bubbling occurred, which was assumed to be simply water vapor release and not CO_2, since there was no evidence of urea and amide formations which should have occurred simultaneously with the CO_2 liberation. The results with the molecular sieves were therefore considered inconclusive.

* Satisfactory storage was only found with sieves containing less than 7% moisture.

In addition to the work done on the isocyanates and water sources, limited investigations (at that time) were being made on polyols (diols and triols), as well as catalysts and surfactants. In general, during this phase only common, commercially available materials were tested. These included liquid diols of equivalent weights of 100 to 200, to insure rigidity, and similar short chain length triols to act as cross linkers. Similarly samples of various commercial catalysts and surfactants were also obtained to be tested in the formulations to be made up.

Because of the poor success achieved in attempting to satisfactorily deactivate the liquid isocyanates by chemical blocking or to utilize encapsulated solid isocyanates and liquid polyols, it was then decided to go to an all solids mixture. Such a system, it was reasoned, should have good storage stability, and if it would melt at a low enough temperature, should be capable of being foamed. The problem then resolved itself to finding solid isocyanates, polyols, catalysts and surfactants of the correct melting point, functionality and vapor pressure to result in a material which would do the following:

a. Be stable when mixed together and stored in a normal atmosphere.

b Melt at a temperature below 200°F (arbitrarily selected)

c. After melting the material should foam up and then

d. Polymerize and become rigid while in the foamed condition.

The first such solid mixture prepared had the formula shown below:

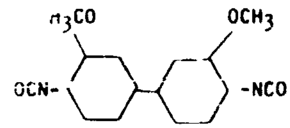
Material	Function	MP °F	Equiv. Weight	Equivalents Used
Diphenyl methane diisocyanate (MDI)	Isocyanate component	99	125	0.5
Bisphenol A	Diol	307	114	0.41
Pyrogallol	Crosslinker	271	42	0.1
Boric Acid	Water source			5% by wt.
Silicone Oil	Surfactant			1% by wt.
8 hydroxy quinoline	Catalyst			5% by wt.

The above formulation resulted in a material which met almost all the requirements. The foam produced, when made in air, was a fair foam, although somewhat friable. When heated in vacuum, however, the vapor pressure of the ingredients (mainly the isocyanate) was so high that as soon

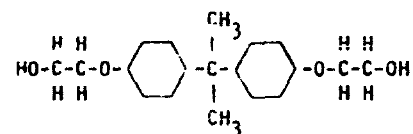
as complete melting occurred the material literally blew itself apart to result in simply a large mass of froth. The importance of this formulation, however, was that it did demonstrate that the solids approach was basically a sound one and that, with improvements, a satisfactory material might be developed along those lines.

Several hundred formulations later such a material was finally developed. The new compound, which used the same type general materials as in the first solids formula, incorporated a higher melting, higher vapor pressure, sterically hindered isocyanate, dianisidine diisocyanate (DADI), as well as a different diol and triol, both melting below 200°F. The structural formulas of the resin components are shown below:

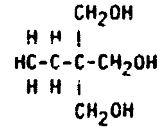
dianisidine diisocyanate (DADI)



p,p'bis (β hydroxy-ethoxy) 2,2' diphenyl propane (diol)

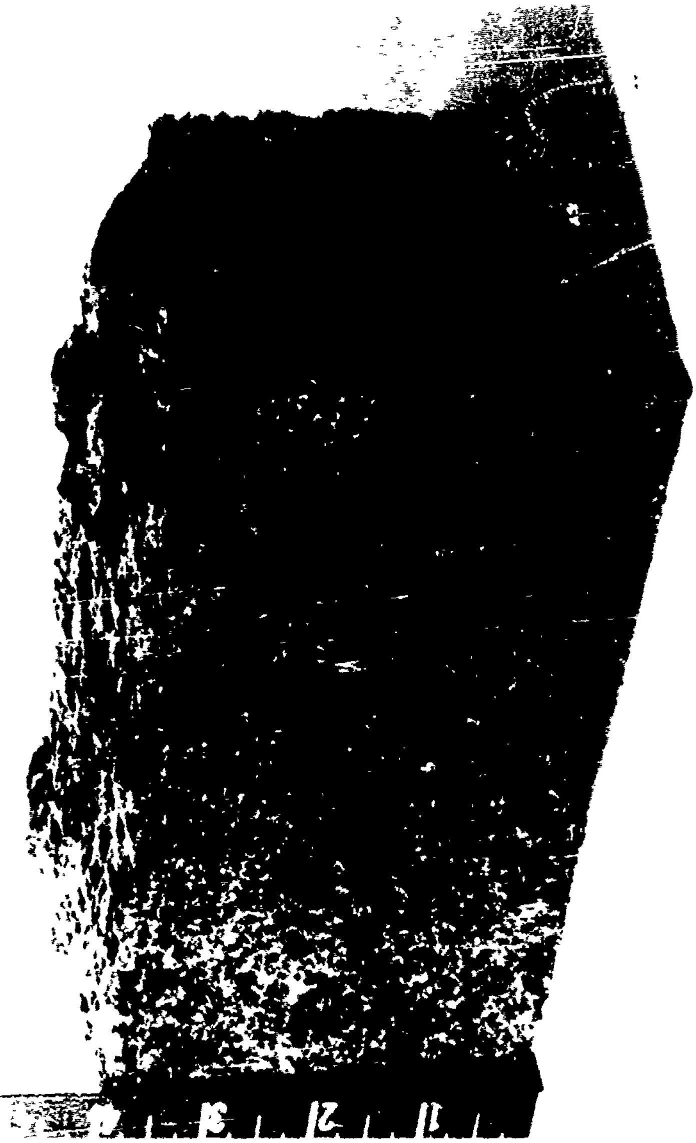


Trimethylol propane (Triol)



In addition, a commercial silicone surfactant, Dow-Corning #113, was used and a commercial catalyst, Metal and Thermit Company T-8. The polyols used were selected after a great number of tests on both commercially available solid polyols and synthesis of a number of polyols at Hughes, the great majority of which turned out to be too high in melting point, or were not solids, or were too long in chain length, etc.

The new compound was a material which melted completely at 175-180°F. It appeared to have an indefinite shelf life when stored at room temperature, although its reactivity apparently is reduced with time. It is not sensitive to normal atmospheric moisture, so needs no special storage conditions. It foamed in a vacuum, without the use of a separate blowing agent (use of vacuum and the vapor pressure of the ingredients turned out to be the key). The resulting foam was 2 to 5 lb. per cu. ft. in density, depending on processing conditons. The compressive strength of the material varies from 15 to 50 psi, depending on density, and on being compressed the material did not shatter. In addition, the foamed material showed surprisingly good strength at 300 to 350°F. By addition of boric acid to the basic mixture a fair foam was obtained at normal pressures. Fig. 1 shows the type of foam made at a pressure corresponding to 160,000 ft. (1 ½ mm Hg approximately).



Even with the production of a satisfactory foaming powder it was found that the development problems were far from over. The diol and triol, though solids, were not hard, crystalline materials, but were somewhat soft and gummy. Therefore, while small amounts of the powder mixture were easily made by hand using a mortar and pestle, attempts to make large amounts by mechanical mixing and grinding always ende up as gummed up messes. A number of techniques were tried including ball mills, a drug mill, a Hobart paddle mixer and a counter current muller. A satisfactory process was finally developed when it was found that by working the material to cause the "gumminess" the temperature would rise and when checked at the right point, a prepolymer would form which was plastic when hot and brittle when cold. This brittle material could then be easily ground up to form the desired powder Using this technique several hundred pounds of powder were prepared for use in fabrication of the required structures.

Fabrication of Demonstration Units

Prior to actually starting fabrication of the balloon and chair it was necessary to develop a method of distributing the powder on the balloon surface. It was also necessary to determine the optimum conditions of time and temperature for forming large amounts of foam, since only small laboratory quantities had been made before. The technique finally adopted for the balloon fabrication utilized the fact that the powder could be pelletized, using a plastic preform press. A number of pellets, 2 3/8 in. diameter by 3/32 in. thick were adhered to the surface of a two foot diameter balloon using Goodyear Plibond cement. Over the pellets was heat sealed (using the pellets as the sealing media) a thin Dacron marquisette cloth, the purpose of which was to prevent running of the material during the period between liquification and final polymerization. Tests on a small (2 ft) balloon indicated a satisfactory structure could be obtained. See Fig. 2

In running a number of tests with various amounts of foam in a vacuum chamber it was found that the maximum height of foam which could be obtained was approximately 4 inches. This limitation was established by the fact that, since the material was endothermic, as the reaction progressed, the developing foam retarded the rate of heat input, so the inner layers of powder would not receive heat as rapidly as the outer layers, and therefore would not foam. Another factor limiting the size of the foam was the fact that after a few minutes of heating, if the powder was too thick, the outer layers would cure or "set" before the inner layers liquified. This cured material then tended to prevent any further expansion from the rest of the mass. It was thus found that approximately 1/4 inch of powder was the limit of material which could be foamed.

With this processing information then it was decided that the large balloon could be fabricated using the adhered-on pellet and restrainer cloth technique. The chair, it was also decided could also be made, but in this case using multiple "blows", to result in the final 2 x 2 x 2 ft. block.

Figure 2. Section of 2 ft. test balloon.

Figure 3 shows the fabrication techniques employed in coating the large 7 ft diameter demonstration balloon. When the balloon was completed, while it was not extremely flexible, it was possible to pack it in a relatively small container for shipment. At Wright-Patterson Air Force Base the balloon was inflated in the vacuum chamber, at a pressure corresponding to approximately 150,000 ft altitude. By suitable pressure regulation the balloon was maintained at 5 inches of water internal pressure during the rigidizing operation. Regidization was accomplished by heating a band approximately one foot wide from pole to pole, as shown in Fig. 4. As the area foamed and rigidized, in approximately 15 minutes, the balloon was rotated to expose a new area. Rigidization was complete in 4 hours. Fig. 5 shows the completed structure.

Fabrication of the chair was conducted in a somewhat different manner. The completely vented mold employed is shown in Fig. 6. Fig. 7 shows the set-up employed for the first "blow", with powder and pellets in place. The chair mold is shown laying on its side so the arms may be formed easier. The heat lamps are directly above the mold. The use of pellets, large and small, was found to be distinctly superior to the use of plain powder. The small pellets on the sides would foam and flow into the depressions left by the main mass of foam as it rose up bubble shaped. (Because of lack of back pressure the material did not follow the mold contours as it foamed). The large pellets were used since it was found they would foam better than plain powder because they were more compact and they helped spread the material more evely than did the powder.

The complete chair was foamed in seven stages, since only approximately 4 inches of foam could be formed at a time. Each layer of reactunts was therefore put in place, foamed, cured and allowed to cool slightly prior to addition of the next layer. Each foam cycle took approximately 20 minutes to complete. However, bringing the chamber to altitude, cooling the foam prior to pressurization, and then adding the new material made each stage take approximately 2 hours to complete. Fig. 8 shows the completed chair.

CONCLUSIONS:

As a result of the preliminary Air Force efforts, and the work reported here, it may be concluded that there are a number of approaches which might be used to produce polyurethane foams in a vacuum environment. These include the solid reactants, encapsulated components, absorbed catalyst-water systems and possibly "blocked" isocyanates. Some of these will require longer research and development programs than others to be fully useable.

The one part, solid reactant heat triggered material so far developed, however, is considered to possess many characteristics which make it ideal for space usage; other than the obvious advantage of elimination of the metering and mixing processes. The material can be triggered by the readily available solar heat. The vacuum environment is used to assist in the blowing process. The powder may be very simply packaged, with no need for vapor tight or pressure resistant containers, refrigeration, etc. Because

Figure 3. Technique for applying foam pellets to balloon surface.

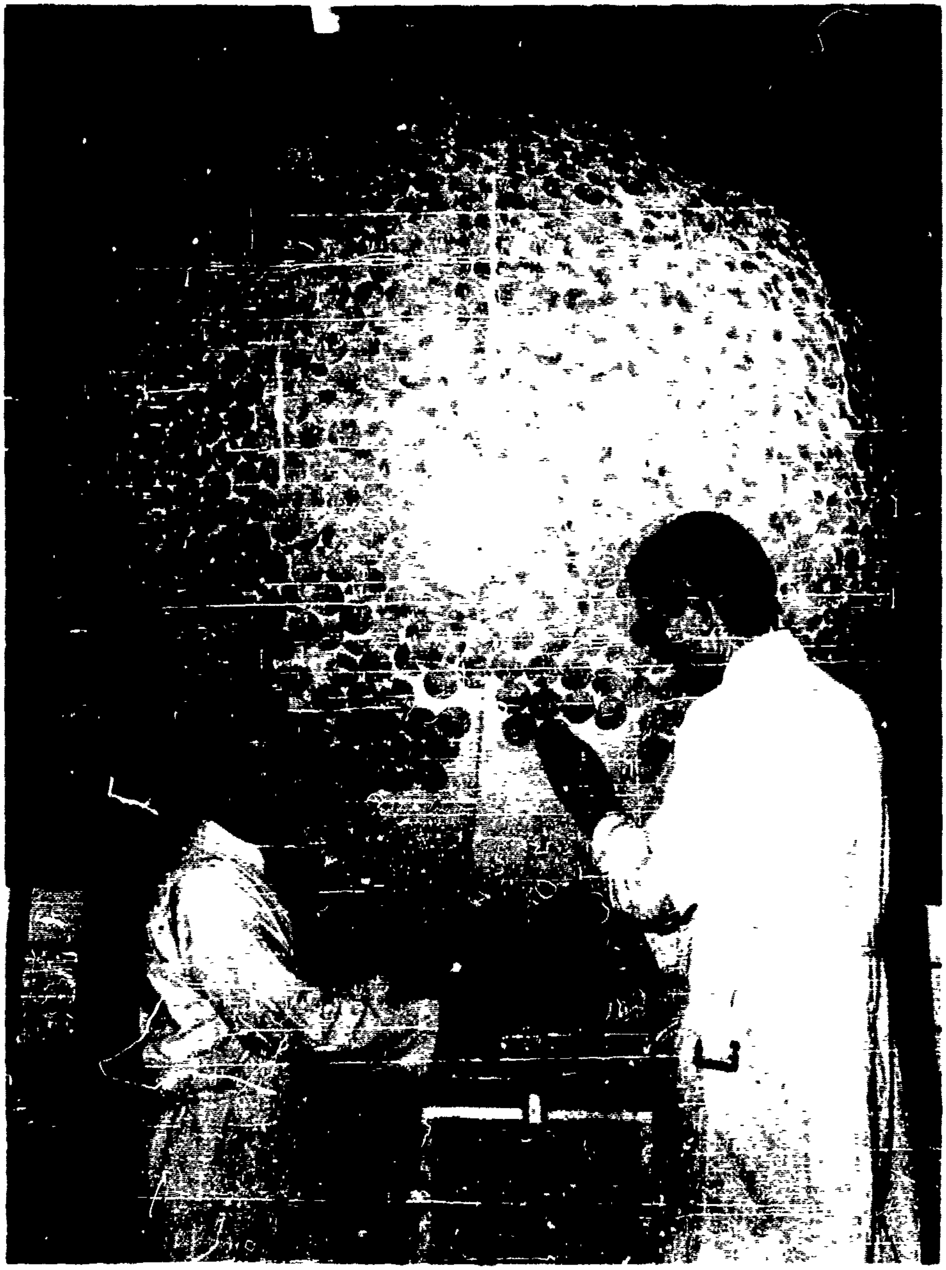


Figure 4. Method used to activate foam on balloon at 160,000 ft.

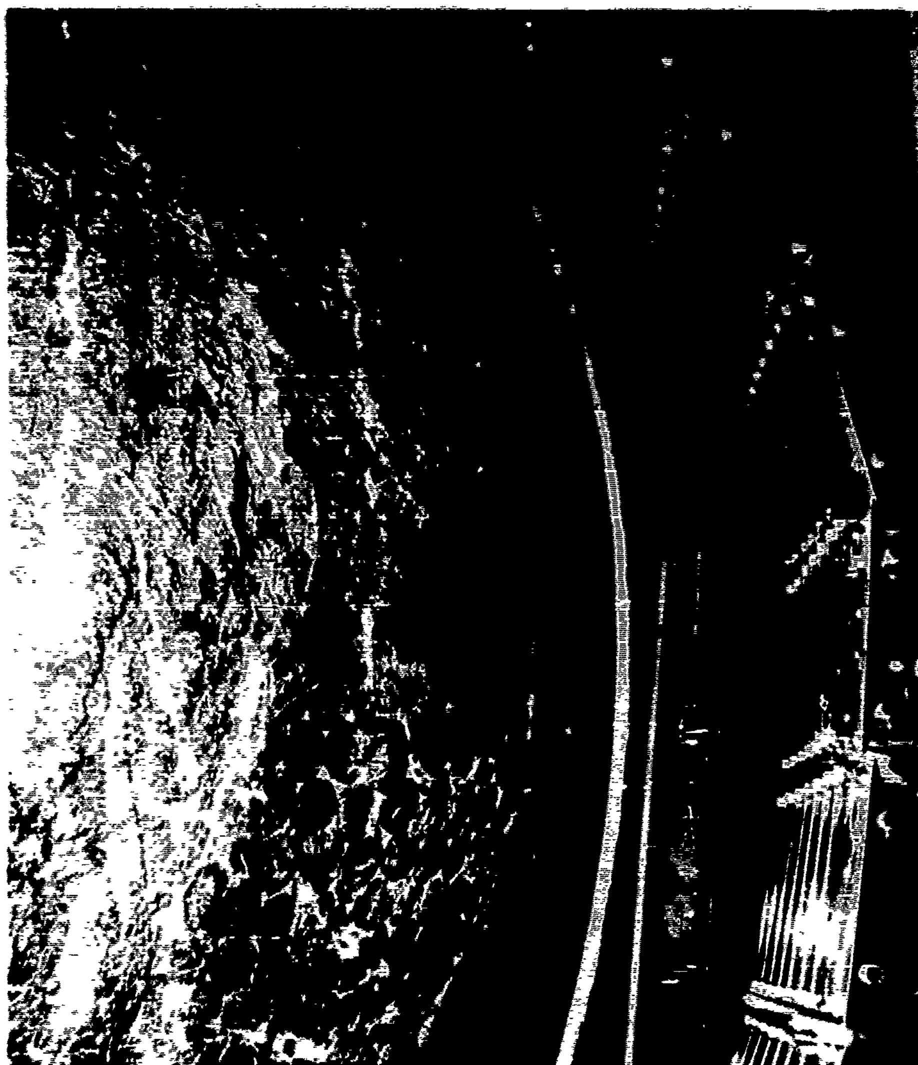
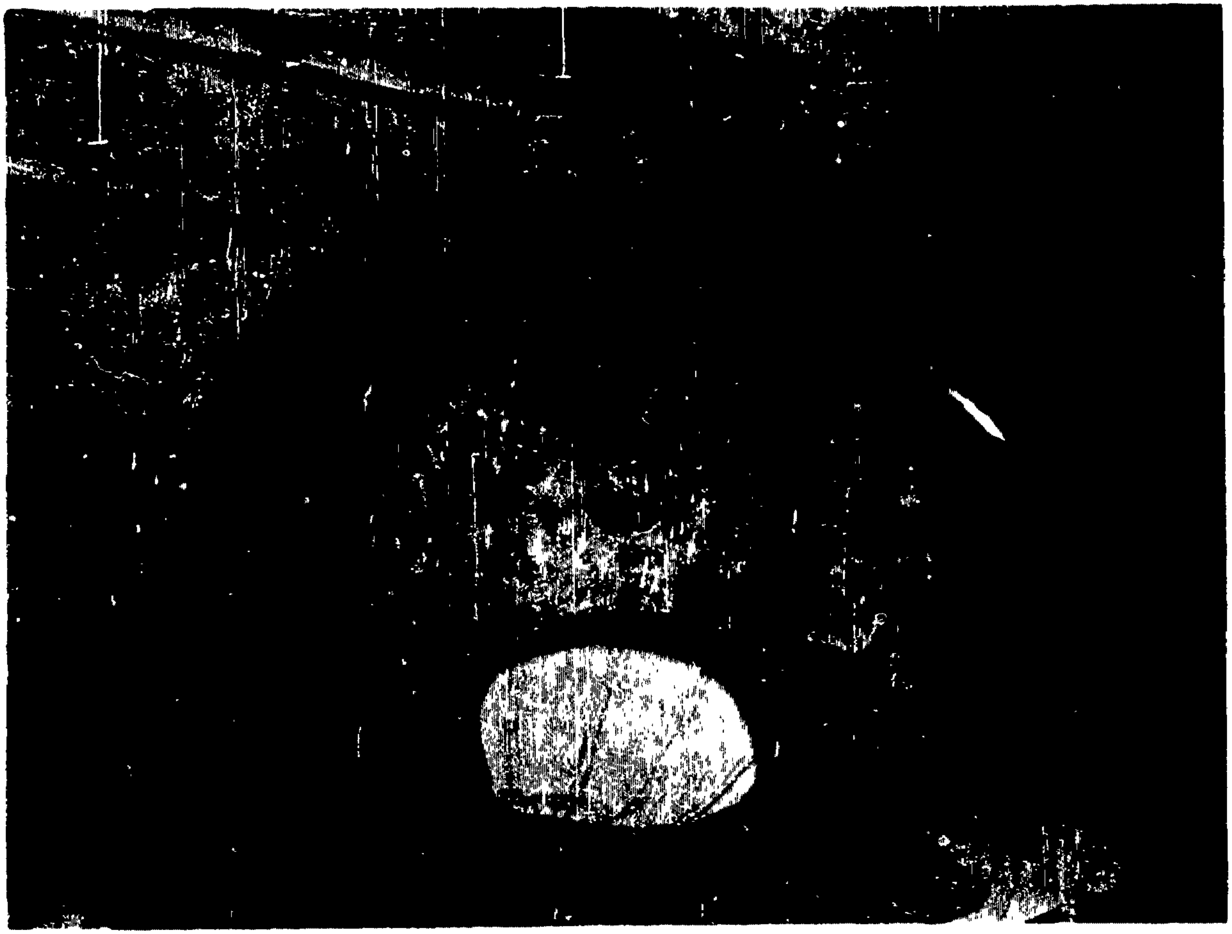


Figure 5. Completed balloon structure.



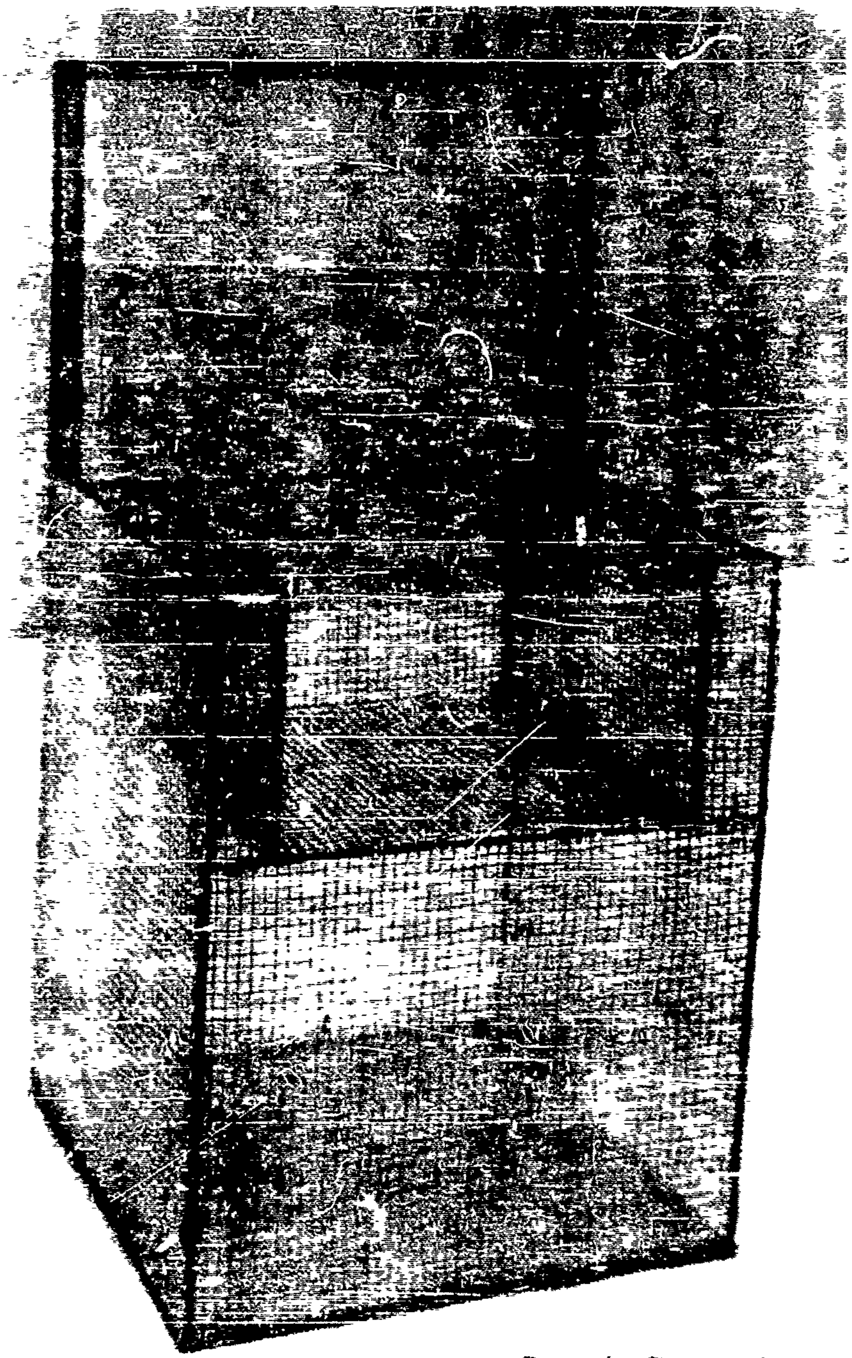


Figure 6. Chair mold.

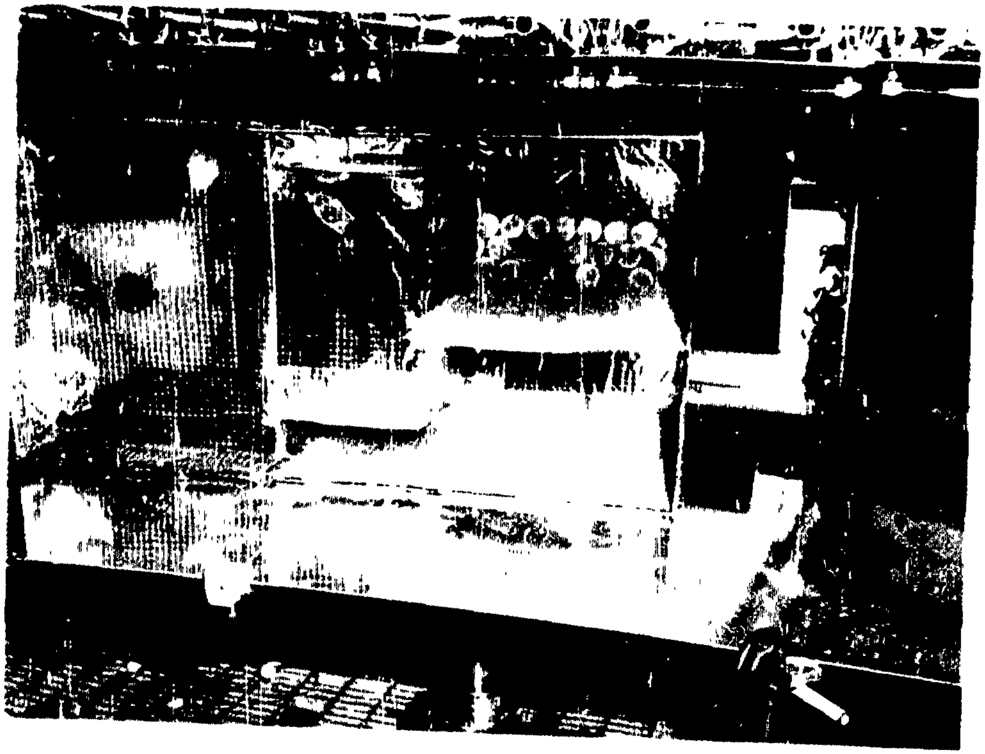


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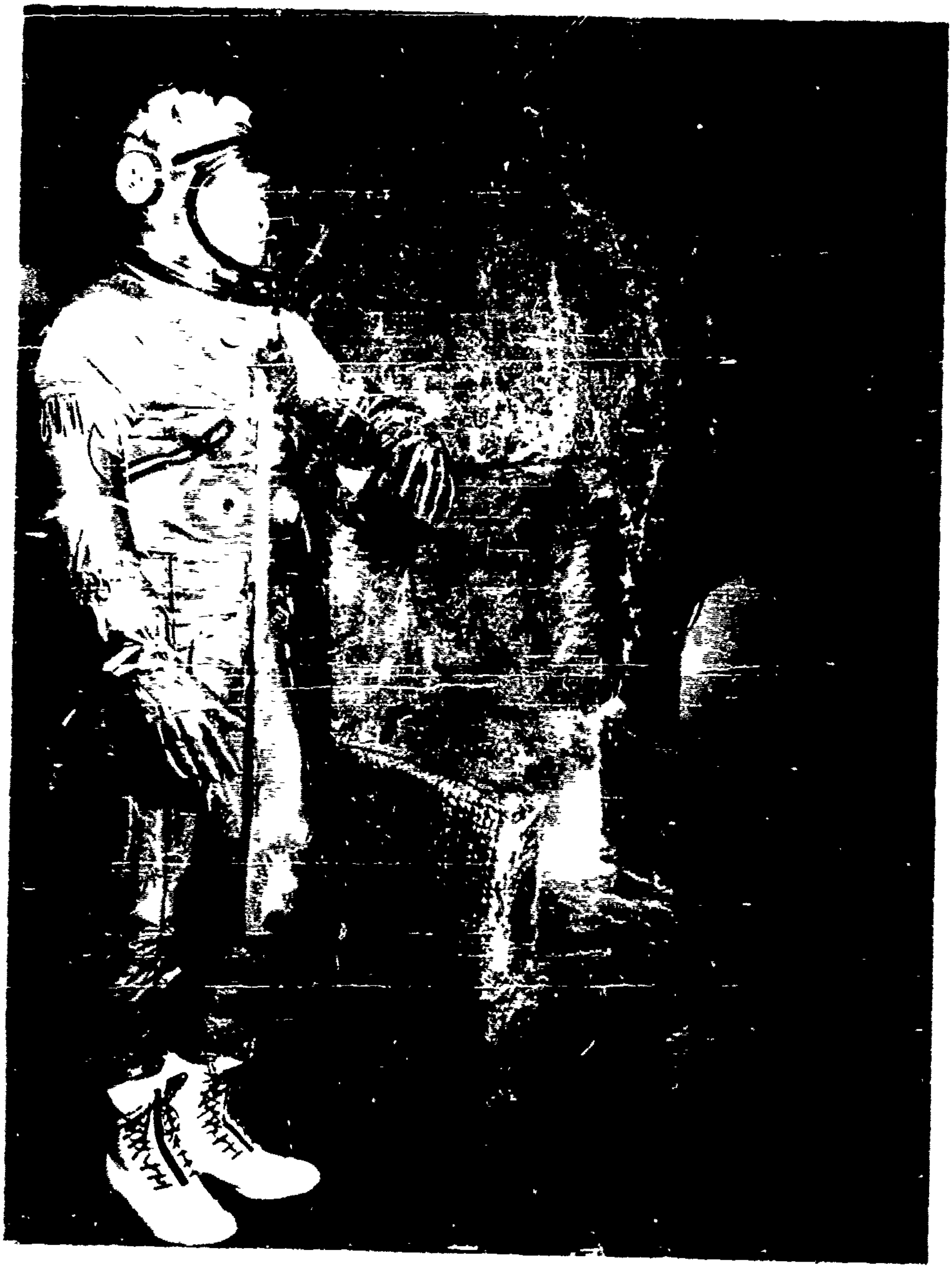
Figure 8. Completed chair.



of the low vapor pressure of the ingredients it is believed that the material has much lower toxicity than conventional polyurethane foamants.

There are also, however, a number of improvements which could be made in the material. One of the most desirable would be a technique to cause the material to exotherm, after initial triggering. In a solid formulation, suggested by the Air Force, such a material has been made. Primarily, the increased rate of reaction is brought about by the use of a more reactive isocyanate, diphenylmethane diisocyanate. Triggering is brought about by a short period of infra-red heating, or by use of electrical heating coils inbedded in the material. Currently, this foamant has a rather limited shelf life, but it does indicate that such an exothermic material is possible.

Another desirable improvement is a technique for spreading the solid reactant foamant evenly over the carrier surface, instead of the pellet attachments. In work which is currently in progress, this has been accomplished on a small scale, and indications are that it can be successfully adapted to large objects or surfaces.

LIST OF ILLUSTRATIONS

MECHANICAL PROPERTIES OF COMPOSITE MATERIALS

FOR EXPANDABLE SPACE STRUCTURES

By Howard L. Price

NASA Langley Research Center
Langley Station, Hampton, Va.

Presented at the Conference on Aerospace Expandable Structures

Dayton, Ohio
October 22-24, 1963

MECHANICAL PROPERTIES OF COMPOSITE MATERIALS FOR EXPANDABLE SPACE STRUCTURES

By Howard L. Price

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ABSTRACT

An analysis is presented of tests to determine the mechanical properties of several composite materials which are used or have potential use in aerospace structures. The composites are approximately 0.001 inch thick, and are laminates of aluminum foil and Mylar film, aluminum foil and polypropylene film, and vapor-deposited aluminum on Mylar film. The analysis is made on the basis of tensile stress-strain, stress-relaxation, and flexural-stiffness tests. The stress-strain and stress-relaxation tests illustrate the mechanical behavior of the composites as influenced by their composition. A comparison is made of the flexural rigidity of the composites as measured by the heavy elastica method.

MECHANICAL PROPERTIES OF COMPOSITE MATERIALS FOR EXPANDABLE SPACE STRUCTURES

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INTRODUCTION

Some of the mechanical and structural applications of materials, on Earth as well as in space, are such that no single material will adequately meet the requirements. As a result, composite materials are often necessary and sometimes offer advantages which are not found in single materials. A composite material is a combination of materials which seeks to utilize certain outstanding properties of each. It has been said (ref. 1) that "the strongest and most efficient materials created by nature . . . and by man . . . have always been composite materials." It is obvious that both strength and efficiency are highly desirable in aerospace structures. Therefore, composite materials can be used to advantage in such structures.

Composite structural materials can be classified into at least three groups. One group is the bonded-layer type of composite, such as laminates. A second group is the matrix type, such as reinforced plastics, and the third group is the atomic structure type which includes the organometallic polymers. This paper will deal with the mechanical properties of some bonded-layer composites, in particular, some aluminum-plastic laminates and an aluminized plastic film. The composites are used or have potential use in aerospace structures, such as passive communications satellites (ref. 2), air-density-measurement satellites (ref. 3), and solar energy concentrators (ref. 4). The mechanical properties were determined by means of tensile tests and stress-relaxation tests, and the flexural rigidity of the composites is measured by the heavy elastica method.

DESCRIPTION OF MATERIALS

The materials are described in table 1 which lists both the nominal thickness and the thickness as measured by an electrically driven micrometer. The measured thickness was used to calculate the cross-section area of the test specimens. Figure 1 shows the cross sections of the composites drawn to approximately the same scale. It is possible, then, to obtain some idea of the relative thicknesses of the materials and their components. The nominal thicknesses are indicated in figure 1.

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*Aerospace Technologist.

TABLE 1.- DESCRIPTION OF COMPOSITE MATERIALS

Material	Nominal thickness, in.	Measured thickness, in.	Weight, lb/in.^{2}	Composition
Echo I	0.0005	0.00042	2.45×10^{-5}	0.5-mil Mylar film with 2,200 Å vapor-deposited aluminum on one side similar to Echo I balloon skin.
A/M Laminate	0.00055	0.00055	3.71×10^{-5}	Two-ply laminate of 0.2-mil aluminum foil and 0.35-mil Mylar film cemented with polyester adhesive.
Echo (A-12)	0.00071	0.0008	5.70×10^{-5}	Three-ply laminate, 0.35-mil Mylar film between 0.2-mil aluminum foil, cemented with polyester adhesive.
A/M/A Laminate	0.00270	0.00285	17.7×10^{-5}	Three-ply laminate, 2-mil Mylar film between 0.35-mil aluminum foil, cemented with polyester adhesive.
Explorer IX	0.002	0.00225	16.4×10^{-5}	Four-ply laminate of 0.5-mil Mylar film and 0.5-mil aluminum foil cemented with polyester adhesive.
X-32B	0.0010	0.0011	3.82×10^{-5}	Three-ply laminate, 0.6-mil polypropylene film between 0.2-mil aluminum foil, cemented with polyester adhesive, 58 percent of aluminum removed in hexagonal pattern by chemical milling.

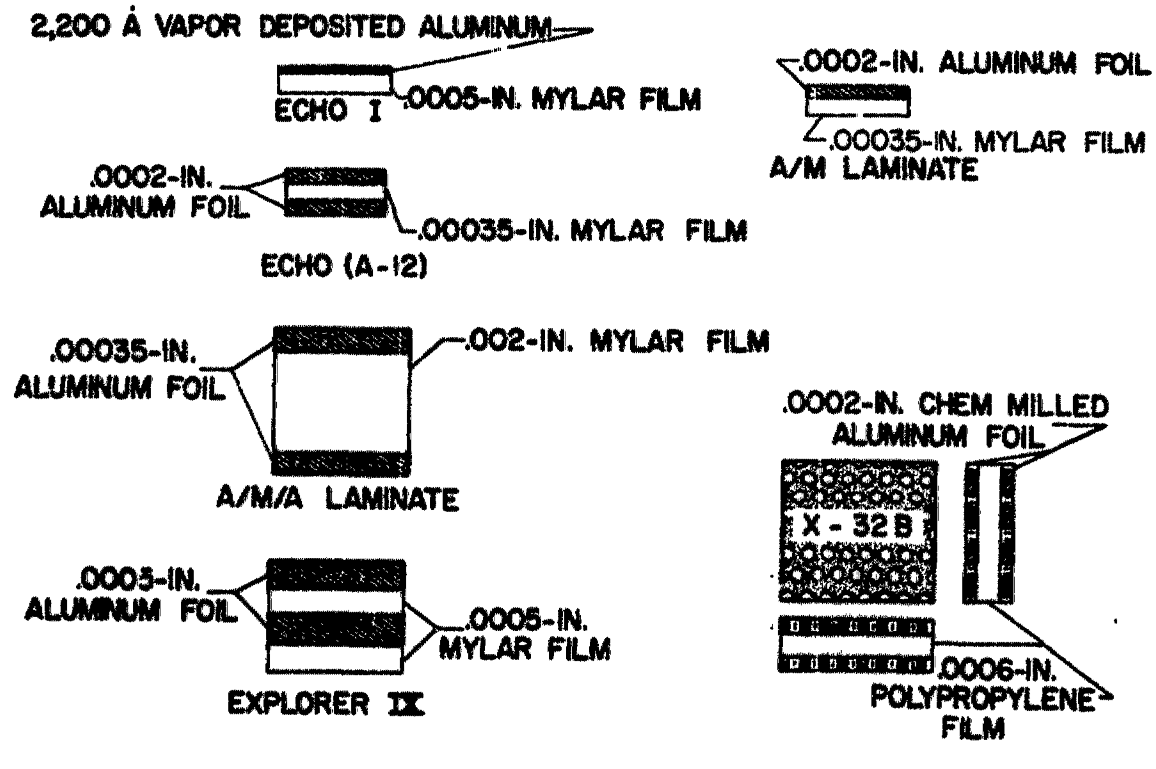


Figure 1.- Cross section of composite materials.

The Echo I material is the type which was used to fabricate the Echo I passive communications satellite. The vapor-deposited aluminum serves as a reflecting surface for radio and radar waves and reduces the degradation by ultraviolet radiation of the Mylar film in the space environment (ref. 5). The Echo (A-12) laminate is the type used to fabricate a 135-foot-diameter sphere. Although it can also be used for communications experiments, the Echo (A-12) balloon was originally designed to demonstrate the feasibility of mechanically rigidizing a large structure in space (ref. 2). The Explorer IX laminate was used to fabricate a 12-foot-diameter air-density-measurement satellite (ref. 3). The Explorer IX satellite has been in orbit since February 1961 and has provided valuable data on the density of the upper atmosphere. Although the aluminum-Mylar (A/M) laminate and the aluminum-Mylar-aluminum (A/M/A) laminate have not been used in space structures it will be shown later that each laminate has certain properties which could be highly useful.

The X-32B laminate was designed to allow a large part of the incident solar radiation to pass through but at the same time to reflect radar waves (ref. 6). Although the solar pressure is quite small (1.3×10^{-9} psi, ref. 7), it can have a considerable effect on large, lightweight, balloon-like structures. For example, solar pressure has helped to alter the orbit of Echo I since the satellite was launched in August 1960 (ref. 8). However, the exposed polypropylene in the X-32B laminate will transmit the solar radiation and the effect of solar pressure on a satellite would be reduced. The etched areas in the aluminum foil are sufficiently small so that the laminate behaves as a solid reflector to radar with a wavelength of at least 3.33 cm.

DESCRIPTION OF TESTS

Three types of tests were used to determine the mechanical behavior of the composite materials - the stress-strain test at constant strain rate, the stress-relaxation test, and the flexural stiffness test.

The stress-strain tests were performed on an Instron Model TT-C testing machine on which the cross-head moves at a constant rate so that the strain rate was constant. The strain was determined by dividing the change in the specimen length by the original distance between the testing machine grips, which was 5 inches. The values of elongation refer to the strain of the test specimen when the specimen failed. The load was measured by a 50-pound capacity load cell, and the area of the test specimen was taken as the product of the width and the total measured thickness of the specimen without regard to the relative areas of metal foil and plastic film.

The stress-relaxation tests also were performed on an Instron testing machine. The test specimen was elongated at a strain rate of 0.04 inch per inch per minute to some predetermined stress. The testing machine cross-head was stopped and observations were made of the continuous relaxation of the stress at the fixed strain for a period of 1,000 minutes (16 hours and 40 minutes).

The flexural rigidity of the composites was measured by the heavy elastica method. The rigidity of the composites is especially important when they are used in large, balloon-like structures which have no load-carrying framework to support the envelope or skin. Satellites in orbit about the Earth are subject to small but continuous deforming loads, such as solar pressure or atmospheric drag (ref. 7). These deforming loads can contribute to the bending or buckling of large areas of unsupported skin. The Echo I, for example, has undergone considerable deformation (ref. 9). The diameter may have decreased (from the original 100 feet) and what appear to be large flat areas have developed in the skin.

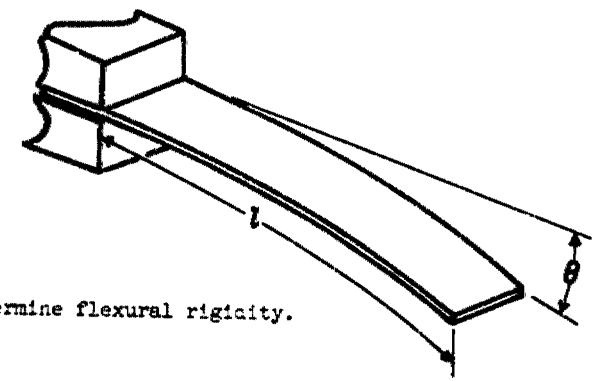
The derivation of the equations for use in the flexural stiffness test may be found in references 10 and 11. A test procedure for measuring the stiffness of fabrics is given in reference 12. In order to measure the rigidity of the composites it was necessary to use a modified test procedure which is presented in reference 13. The present tests were performed by projecting a strip of material from a horizontal surface (fig. 2(a)). Measurements were made of the length of the overhang l and the deflection θ of the free end below the horizontal. By entering the curve in figure 2(b) with the measured value of θ, the ratio c/l can be determined, where c is referred to as bending length. The flexural rigidity, then, is simply the product of c^3 and the weight of the strip per unit length.

RESULTS AND DISCUSSION

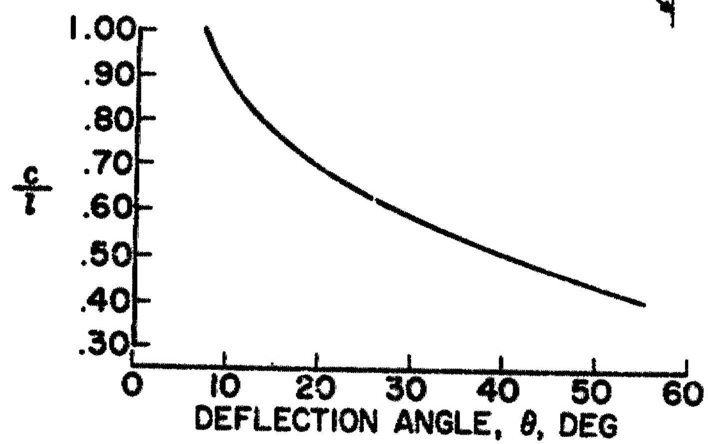
The tensile properties and flexural rigidity of the composites are listed in tables 2 and 3 and illustrated in figures 3 to 6. These data were obtained from references 13 and 14 for 1/2-inch-wide strips of material and have been normalized to unit width.

Tensile Properties

The tensile properties of the composites are listed in table 2 which includes the values of the tensile strength, Young's modulus, and elongation at break. The Echo I material has the highest tensile strength (27,000 psi) and the A/M laminate has the lowest (11,600 psi). This low value is the result of the comparatively early failure of the aluminum foil when the laminate is being elongated. When the aluminum foil breaks, the laminate is considered to have failed even though the plastic film may be intact. In the case of homogeneous (that is noncomposite) materials the tensile strength is a valid indication of the load that the material can carry. In comparing composite materials, however, the tensile strength may not be indicative of the load-carrying ability, especially when the total thickness is used to compute the cross-section area. Therefore, table 2 includes the values of the load which is required to break a 1-inch-wide strip of material. The A/M/A laminate has the highest breaking load (61.9 pounds) and the A/M laminate has the lowest (6.4 pounds) of the composites which were tested. The ranking of the six composites according to tensile strength and according to breaking load results in a different sequence, with only the A/M laminate and the X-32B laminate retaining their same position in each sequence.



(a) Measurements required to determine flexural rigidity.



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(b) Relationship between deflection angle θ, length of overhang l, and bending length c.

Figure 2.- The heavy elastica method of measuring flexural rigidity.

TABLE 2.- TENSILE PROPERTIES OF 1-INCH-WIDE STRIPS

OF COMPOSITE MATERIALS

Material	Tensile strength, psi	Breaking load, lb	Young's modulus, psi	Extensional stiffness, lb/in.	Elongation
Echo I	27,000	11.3	0.66×10^6	2.77×10^2	158
A/M Laminate	11,600	6.4	0.53×10^6	2.92×10^2	11
Echo (A-12)	13,700	11.0	2.73×10^6	21.84×10^2	12
A/M/A Laminate	21,700	61.9	2.08×10^6	59.28×10^2	149
Explorer IX	13,000	29.3	3.68×10^6	82.80×10^2	26
X-32B	------	14.7	-----------	2.85×10^2	43

TABLE 3.- FLEXURAL RIGIDITY OF 1-INCH-WIDE STRIPS OF COMPOSITE MATERIALS

AS MEASURED BY THE HEAVY ELASTICA METHOD

Material	Flexural rigidity (EI), lb-in.^{2}	Rigidity relative to Echo I	Weight efficiency, in.^{4}	Thickness efficiency, lb-in.	Remarks
Echo I	5.06×10^{-6}	1	2.1×10^{-1}	6.8×10^{4}	Aluminized side in tension
A/M Laminate	6.67×10^{-6}	1.3	1.8×10^{-1}	4.02×10^{4}	Rigidity calculated assuming aluminum foil effective in bending
Echo (A-12)	5.84×10^{-4}	115	1.03×10	1.14×10^{6}	
A/M/A Laminate	5.02×10^{-3}	992	2.84×10	1.76×10^{5}	
Explorer IX	4.56×10^{-3}	901	2.8×10	4.0×10^{5}	Aluminum side in tension
X-32B	2.42×10^{-4}	48	6.3	1.82×10^{5}	

Table 2 includes the values of Young's modulus which were determined on the basis of total cross-section area. The Young's modulus of the Explorer IX laminate (3.68×10^6 psi) is approximately seven times that of the Echo I material (0.66×10^6 psi). Of more practical interest, however, is the extensional stiffness. The extensional stiffness can be thought of either as the product of Young's modulus and the material thickness, or as the force required to extend a unit width strip of material to twice its original length (i.e., 100-percent strain). The extensional stiffness of the composites is included in table 2. The Echo I material, the A/M laminate, and the X-32B laminate all have an extensional stiffness of nearly 3 pounds per inch. The Explorer IX laminate has the highest value (82.8 pounds per inch) of extensional stiffness of the composites.

Although the extensional stiffness is given in pounds per inch, it does not follow that extending the material to 100-percent strain will result in the force shown in the extensional stiffness column. The reason, of course, is that the composite may yield or even break at much smaller values of strain. However, the extensional stiffness provides a reliable means of evaluating the force required to strain composites of widely different composition.

The elongation at break requires no change for composite materials because the cross-section area does not enter into the calculation as it does for the tensile strength and Young's modulus. Both the Echo I material and the A/M/A laminate have high elongations, on the order of 150 percent. For a test specimen with an original length between the grips of 5 inches, the length at break was about 12.5 inches. The high elongation reflects the large amount of Mylar film which the composites contain. The elongation of the A/M, the Echo (A-12), and the Explorer IX laminates are the strains at which the aluminum foil fails. In general, the Mylar film remains intact.

It is concluded, then, that the A/M/A laminate can withstand the largest tensile load, the Explorer IX laminate has the highest extensional stiffness, and the Echo I material has the highest elongation of the materials which were tested. Furthermore, it can be very misleading to describe the tensile properties of composites by means of the usual tensile strength and Young's modulus values based on total cross-section area. It is more accurate, especially when comparing different composite materials, to present the tensile properties in terms of breaking load, extensional stiffness, and elongation.

Calculated Stress-Strain Values

When a composite material is stressed, the components are not, in general, under the same stress. It is instructive to determine what stresses the components experience so that, if the properties of the components are known, a composite can be constructed which will have certain predetermined properties. Therefore, the stress at selected strains was calculated for the Echo (A-12) laminate and the results are shown in figure 3.

The experimentally determined stress-strain curves are shown in figure 3 for the aluminum foil and Mylar film which is used in the Echo (A-12)

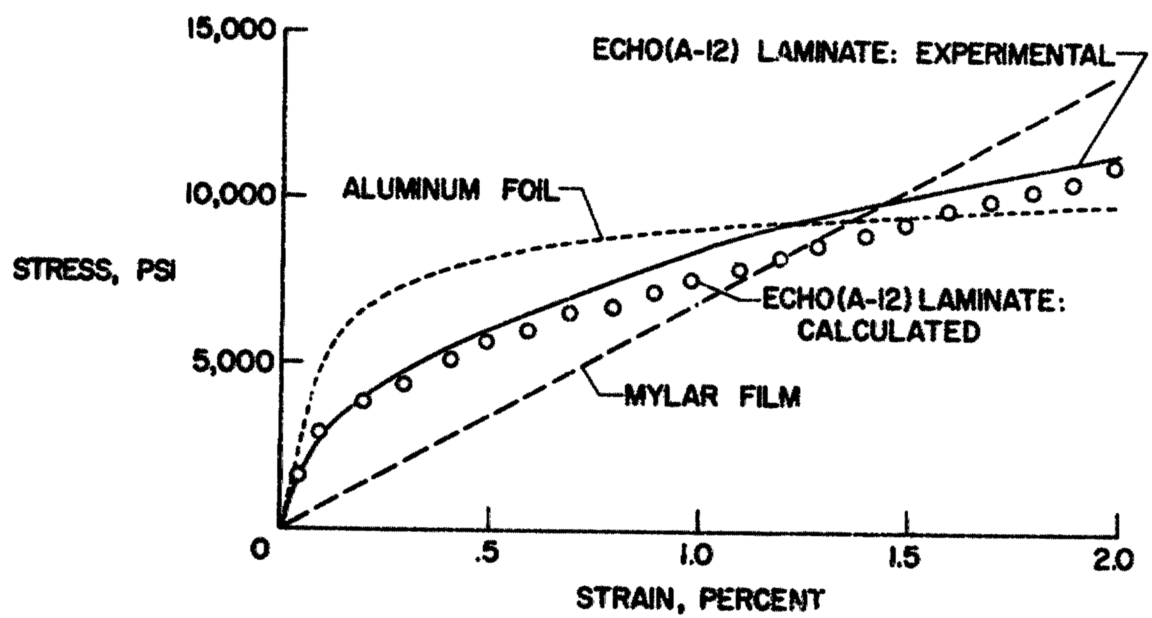

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Figure 3.- Experimental and calculated stress-strain values of Echo (A-12) laminate.

laminate. The curves were obtained from tests of 1/2-inch-wide strips of material which were loaded at a strain rate of 0.04 inch per inch per minute. In calculating the stress on the composite it is assumed that the composite load is the sum of the loads on the components (ref. 15). In addition, it is assumed that the aluminum and the Mylar film undergo equal strain in the laminate so there is no slippage between the layers. For a given value of strain the load on each component is determined from the stress-strain curve. The component loads are added and their sum is divided by the area of the laminate, thereby providing the calculated laminate stress for the given strain.

The calculated stress-strain values are compared with the experimental stress-strain curve for the Echo (A-12) laminate in figure 3. There is fairly good agreement for strains up to about 0.5 percent which is beyond the yield stress of the laminate. At higher strains the calculations indicate a lower stress than those that were actually measured. However, the difference is rather small (about 600 to 700 psi) and fairly constant, suggesting that the Young's modulus for the Mylar film might be higher (0.79×10^6 psi) than the value of 0.69×10^6 psi which was used in the calculations. It is concluded, then, that it is valid to combine the stress-strain curves of the components in order to obtain the stress-strain curve of the laminate.

A comparison of the experimental curves (fig. 3) of the components and the Echo (A-12) laminate shows that, for strains up to about 1.5 percent, the aluminum-foil stress is higher and the Mylar film stress is lower than the laminate stress. At higher strains, or laminate stress above approximately 10,000 psi, the Mylar is more highly stressed than the laminate. Furthermore, a laminate stress of 4,000 to 5,000 psi is required to yield the aluminum foil. Once the aluminum foil has passed its yield stress it will take a permanent set, although the Mylar film will still be elastic. When the laminate is released, the Mylar film will attempt to recover its original, undeformed length. Complete recovery will be prevented by the permanent deformation of the aluminum foil. Therefore, the aluminum will be in compression and the Mylar film will be in tension after the laminate stress is released.

The tension and compression in the components has no apparent effect on the Echo (A-12) laminate. It can be seen from the cross section of the laminate (fig. 1) that the tension force in the Mylar film is balanced by the compression forces in the aluminum on each side of the film. However, in the Echo I material, the A/M laminate, and the Explorer IX laminate the forces are not balanced and the resulting torques deform the material.

In order to illustrate the effect of the torques, 1/2-inch-wide strips of the composites were subjected to several tensile loads and the resulting deformation is shown in figure 4. As the load increases so does the curvature of the composites. The small sections which were cut from the test strips tended to curl into small cylinders. The edges of the test strip of the Echo I material curled after 4 pounds of load so the strip itself did not deform. After 5 pounds of load the vapor-deposited aluminum was so thin that there was no appreciable deformation as shown by the section that was cut from the test strip. Although the torques are small (an estimated 2.5×10^{-4} inch-pound for the A/M laminate subjected to a $1\frac{1}{2}$-pound load)

they may become significant in a null gravity field and cause undesirable deformations. On the other hand, such torques might prove to be useful as a restraining or restoring force in a simple spring or damping system.

Stress Relaxation

The effect of loading a laminate until the aluminum has exceeded its yield stress has been demonstrated above and in figure 4. In the case of a laminate with balanced forces, the yielding of the aluminum is indicated by a decrease in the slope of the stress-strain curve (fig. 3). A further indication of the effect is obtained by means of the tensile stress-relaxation test.

The representative stress-relaxation curves for the Echo (A-12) laminate are shown in figure 5 where the stress is plotted against the logarithm of the time in minutes. The tests were conducted with initial laminate stresses in a lower range of 1,000 to 5,000 psi, and with stress in a higher range of 6,000 to 10,000 psi (see ref. 14). At all stress levels the stress decreases approximately linearly with the logarithm of time from 0.1 minute to 1,000 minutes (16 hours and 40 minutes). If the curves are extended they intersect at two points. The curves for the initial stress in the lower stress range intersect at a stress of about 100 psi and a logarithm of time of 17.7. The curves for the higher range stresses intersect at a stress of 1,300 psi and a logarithm of time of 14.8.

The two points of intersection reflect the dual response of the laminate to long time, fixed deformation. For laminate stresses up to 4,000 psi the aluminum foil is still below its yield stress and the Mylar film is at a comparatively low stress of less than 1,500 psi. The aluminum, then, carries most of the load on the laminate so the curves in the lower stress range are virtually those of the aluminum. At the higher stresses the aluminum has exceeded its yield stress and the Mylar film may be loaded to a stress as high as 10,000 psi. Therefore, the laminate undergoes a more rapid rate of stress reduction in the upper stress range than it does in the lower. The difference, however, is rather small. In the lower range the stress decays at the rate of about 4.8 percent of the original stress per decade of time and in the upper range the rate is 6 percent. As a result, if the laminate is fabricated into a structure in such a way that stress gradients occur across the surface of the laminate (gradients caused by such elements as seams, pole caps, or local reinforcements) it is unlikely that the stress gradient will decrease to any practical degree. For example, for an original laminate stress of 10,000 psi, more than 100 years would be required for the laminate to relax to 5,000 psi (fig. 5). It can, of course, be very misleading to extrapolate time-dependent data. The above example, however, although it involves such extrapolation, does give some idea of the long times which would be required for the Echo (A-12) laminate to undergo extensive stress relaxation.

Flexural Rigidity

The flexural rigidity of the composites is listed in table 3 and illustrated in figure 6. The flexural rigidity is the product of Young's modulus,

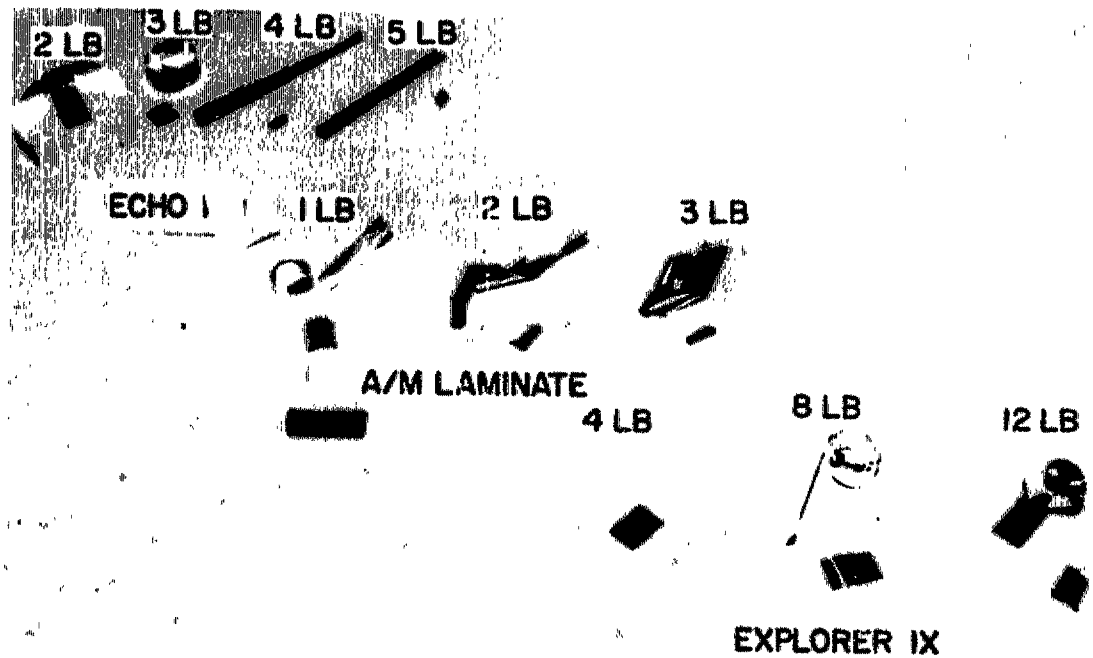

Figure 4.- Deformed condition of Echo I material, A/M laminate, and Explorer IX laminate after having been subjected to the indicated loads.

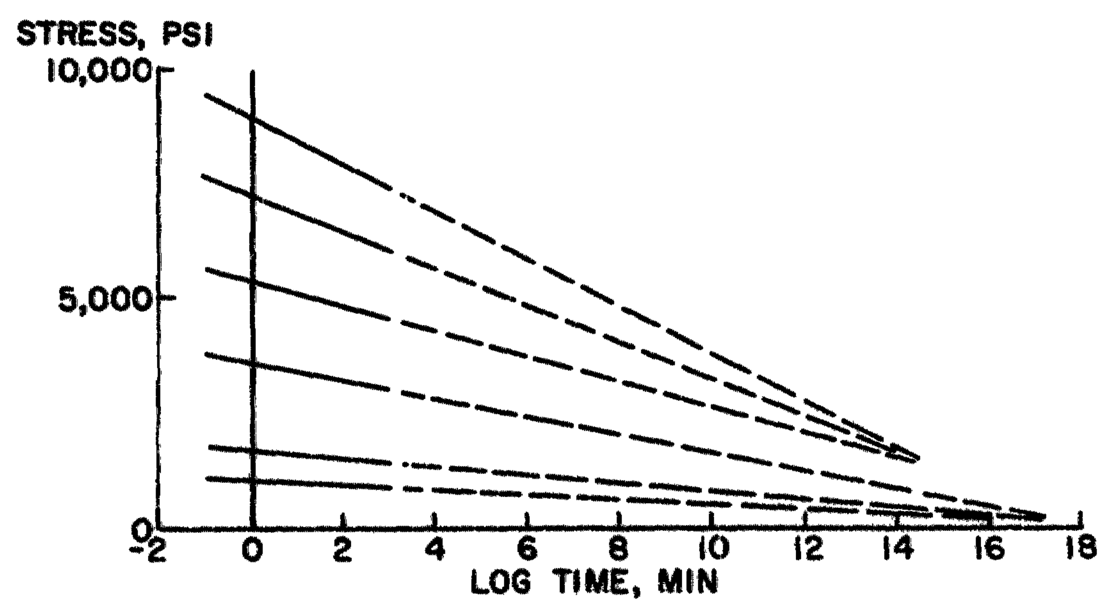

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Figure 9.- Tensile stress relaxation of Echo (A-12) laminate.

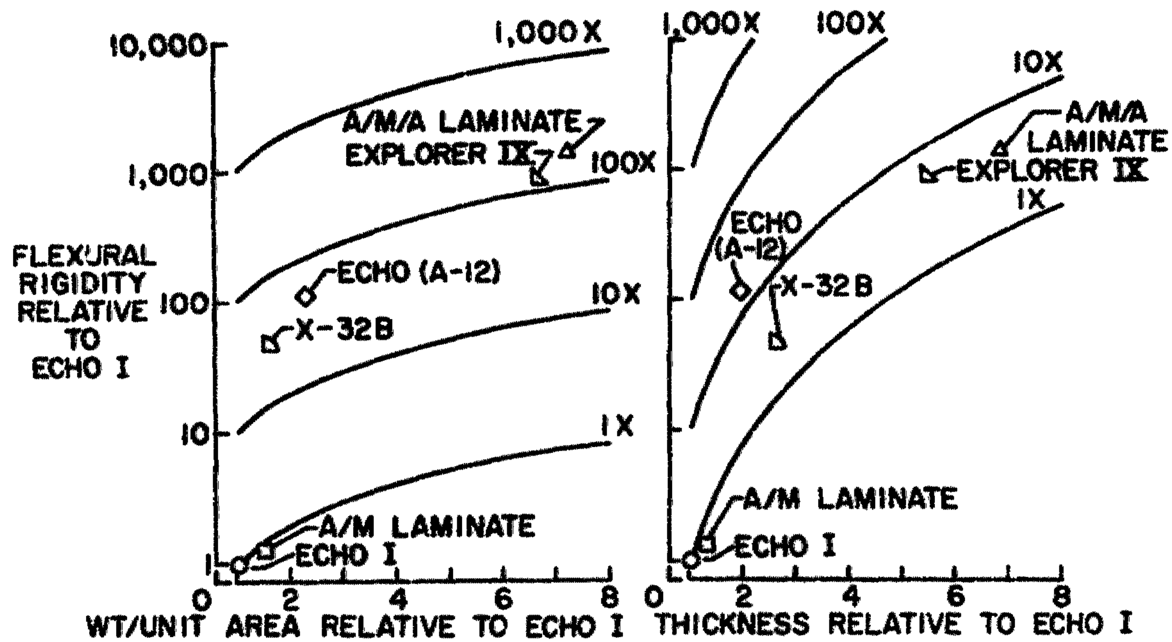

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Figure 6.- Weight and thickness efficiencies of composite materials relative to Echo I material.

E, and I, the moment of inertia for flexure. Because of its tendency to curl it was impossible to measure the flexural rigidity of the A/M laminate in the same manner as the other composites was measured. Therefore, the flexural rigidity of the laminate was calculated.

All of the flexural rigidity values are quite low for structural purposes. They fall in the range of 10^{-6} to 10^{-4} lb-in.^{2} which is small in comparison to the flexural rigidity of approximately 8×10^{-1} lb-in.^{2} for 0.01-inch-thick aluminum. The lowest value of rigidity, 5.06×10^{6} lb-in.^{2}, is that of the Echo I material. The A/M/A laminate has the highest rigidity, 5.02×10^{-3} lb-in.^{2}. The rigidities relative to the Echo I material are listed in table 3. The Echo (A-12) laminate, which was designed to be stiffer than the Echo I material, is over 100 times as rigid. The ratio of the rigidity of the Echo I material to the Echo (A-12) laminate to the X-32B laminate is 1/115/48. The ratio given in reference 6 for the same composites is approximately 1/57/5. The latter ratio, based on design calculations, is more conservative than the ratio which was determined by the flexural rigidity tests. The difference between the two ratios gives some indication of how difficult it is to obtain agreement between the calculated and the measured flexural rigidities for thin composite materials.

Increased resistance to bending may or may not be an advantage, depending on the application of the composite. If the increased rigidity is accompanied by a substantial weight increase, then the more rigid composite may not be as efficient as one which is less rigid. The flexural rigidity of the composite, divided by its weight per unit area, is taken as the weight efficiency. The values are listed in table 3 and are equal to c^3, the value of c being determined from the curve in figure 2b. In reference 10 the value c is referred to as the bending length and is considered to be a measure of the quality of a fabric. As it is used here c is a measure of the efficiency of a composite material in resisting flexure.

A higher rigidity may be obtained by increasing the thickness of the material. In a homogeneous material the flexural rigidity increases as the cube of the thickness. A thicker material, however, requires a larger volume for packaging and may present some folding problems. Therefore, the flexural rigidity of the composite, divided by its measured thickness, is taken as the thickness efficiency and the values are listed in table 3.

The A/M laminate has the lowest $(1.8 \times 10^{-1} \text{ in.}^4)$ and the A/M/A laminate has the highest $(2.84 \times 10 \text{ in.}^4)$ weight efficiency of the composites which were tested. The flexural rigidity of the A/M laminate may be lower than the true value because the rigidity was calculated on the assumption that only the aluminum is effective in bending. Even if the calculated rigidity were in error by as much as 20 percent, the weight efficiency would be comparable to the efficiency of the Echo I material. From the standpoint of thickness, the A/M laminate has a low efficiency of 4.02×10^4 lb-in. The Echo (A-12) laminate has a high thickness efficiency of 1.14×10^6 lb-in.

A comparison of the flexural rigidity, the weight, and the thickness of the composites relative to the Echo I material is shown in figure 6. A logarithmic scale is used for the relative flexural rigidities in order to

accommodate the wide spread in the values. Included in figure 6 is a set of curves which show the flexural rigidity which would be equal to a weight or thickness efficiency of 1, 10, 100, and 1,000 times the efficiency of the Echo I material. For example, a composite that is 50 times as rigid but which weighs five times as much would be located on the 10X curve because it would have ten times the weight efficiency of Echo I.

It can be seen in figure 6a that the weight efficiency of the A/M laminate is lower than that of the Echo I material. The highest (135X) efficiency is achieved by the A/M/A laminate followed closely by the Explorer IX laminate which has a weight efficiency that is 133 times that of the Echo I material. The high efficiencies are obtained at a considerable increase in weight of 7 to 8 times the weight of the Echo I material. However, the laminates should prove to be useful in cases in which a small (and, therefore, low total weight) but highly rigid structure is required. The requirements of size and rigidity were in fact encountered in the Explorer IX air-density-measurement satellite (ref. 3). The Explorer IX laminate, then, represents a case in which a laminate was designed for a particular application and then satisfactorily employed for it.

Both the Echo (A-12) laminate and the X-32B laminate have higher weight efficiencies (49 and 30 times) than the Echo I material. The X-32B may be more efficient in orbit than is indicated by figure 6. The reason is that in the flexural rigidity test the gravitational force or weight of the material is the deforming load. The chemical milling of the X-32B laminate removes 58 percent of the aluminum, and relieves the laminate of approximately one-half of the deforming load of solar pressure. The polypropylene windows contribute to the weight which deforms the material in the flexural rigidity test. These same windows would contribute a negligible load under the action of solar pressure. Therefore, the X-32B laminate may be more efficient than the flexural rigidity test indicates.

The thickness efficiencies of the composites are illustrated in figure 6b where it is shown that A/M laminate has the lowest efficiency. The A/M/A, X-32B, and Explorer IX laminates have thickness efficiencies which are approximately 2, 3, and 6 times that of the Echo I material. The Echo (A-12) laminate has the highest efficiency (17 times) and the lowest thickness (2 times) relative to Echo I material. The combination of low thickness and high efficiency indicates that the Echo (A-12) is suitable for large structures, such as the Echo (A-12) satellite, or for large structures in which rigidity is required but packaging space is limited.

The above discussion indicates increased efficiency in resisting bending is accompanied by an increase of weight and/or thickness of the composite. An increase in weight or thickness does not necessarily bring about an increase in efficiency, however. The A/M laminate is an example. It is not good enough, then, to design or use laminates which are heavier or thicker than Echo I material, even though they may have a higher flexural rigidity. The candidate laminate might be no more efficient than a scaled-up version of the Echo I material. On the other hand, an Echo I type material of the thickness of the Explorer IX laminate might present some problems in folding and packaging. As a result, a laminate incorporating aluminum foil might be necessary so that the required flexibility could be achieved, even at the loss of some of the efficiency. A composite material for use in

balloon-like structures, then, represents a compromise among the factors of weight, thickness, rigidity, and flexibility.

CONCLUDING REMARKS

The mechanical properties have been determined of six composite materials which have been used or have potential use in aerospace structures.

It has been found that, in comparing the tensile properties of composite materials, it is more accurate to express the properties in terms of breaking load and extensional stiffness rather than strength and Young's modulus. The stress-strain or load-strain curve of a composite can be estimated if the stress-strain curves of the components are known. In the aluminum-Mylar laminates the residual stresses in the components may act in opposite directions. As a result, these forces can severely deform a laminate in which the forces are not balanced. The dual response to long time, fixed strain of a two component laminate has been demonstrated by the stress-relaxation test. For laminate stresses below those at which the aluminum has reached yield, the rate of stress decay is lower than the case in which the aluminum has yielded.

The flexural rigidity of the composites has been measured by the heavy elastica method. It has been found that increased rigidity is obtained at the cost of increased weight and/or thickness.

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Figure 1.- Cross section of composite materials.

(a) Measurements required to determine flexural rigidity.

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(b) Relationship between deflection angle θ, length of overhang l, and bending length c.

Figure 2.- The heavy elastica method of measuring flexural rigidity.

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Figure 3.- Experimental and calculated stress-strain values of Echo (A-12) laminate.

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Figure 4.- Deformed condition of Echo I material, A/M laminate, and Explorer IX laminate after having been subjected to the indicated loads.

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Figure 5.- Tensile stress relaxation of Echo (A-12) laminate.

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Figure 6.- Weight and thickness efficiencies of composite materials relative to Echo I material.

EXPANDABLE DRAG DEVICES FOR MACH 10 FLIGHT REGIME

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Space and the high-speed, high-altitude flight associated with it requires new methods of stabilization and deceleration for the recovery of such payloads as manned escape capsules, personnel ejected from space capsules, rocket boosters, nose cones and instrument data packages. Initial stabilization is required so that protective re-entry devices (heat shields, ablation shields, drag producing devices) of a payload tumbling or disoriented in space can be aligned with the flight path. Initial deceleration is required to reduce aerodynamic heating and loading and to reduce the velocity of the payload gradually in a varying dynamic loading regime. This is done essentially by decreasing the ballistic coefficient, W/C_DA.

These new stabilization and deceleration methods will provide for a controlled descent over a known trajectory and, coupled with acquisition aids, will enable a successful recovery within a predetermined area.

If conventional recovery methods are to be utilized in the final stage, it is particularly important that the velocity of the payload be reduced gradually from high-speed, high-altitude conditions. Recent investigations indicate that conventional parachutes are not satisfactory for this first-stage deceleration because of aerodynamic heating and erratic loading at supersonic flow conditions. Under Contract AF33(616)-6010, Goodyear Aerospace Corporation studied the feasibility of inflatable fabric structures for recovery systems for speeds to Mach 4 and altitudes to 200,000 feet. This program included theoretical analysis as well as laboratory, wind tunnel, free-fall, and rocket powered flight tests. Under Contract AF33(616)-8015, the company investigated various shapes of inflatable decelerators for operation at Mach 10, altitudes to 200,000 feet and temperatures to 1500 F. The investigation also included study of drag area variation during descent to eliminate final stage parachute recovery. Deployable decelerators were investigated analytically and tested in wind tunnels at the above aerodynamic conditions.

In June, 1963, Goodyear Aerospace was awarded by the Aeronautical Systems Division of the USAF, the prime contract for the program designated as ADDPEP (Aerodynamic Deployable Decelerator Performance Evaluation Program) (Contract AF33(657)-10955). This program, in essence, includes theoretical investigation, evaluation, tests and modification of newly-designed deployable decelerators categorized in three groups: (1) hi-q parachutes, which are large ribbon-type parachutes needed for large payload recoveries at transonic and supersonic speeds at high dynamic pressures; (2) small supersonic parachutes capable of stable and reasonable high-drag performance at Mach numbers up to 5; and (3) the investigation, evaluation and modification of balloon-type (Ballute) decelerators capable of stable and high-drag performance up to Mach numbers of 10.

This program entails the design modification and manufacture of two new basic type rocket-boosted vehicles. These vehicles are test devices weighing 400 to 2100 pounds, depending upon the application, and are capable of monitoring performance of the decelerators during their flight. The vehicles may be boosted by a large number of various solid booster combinations to the desired trajectories and test points. The performance envelope of these vehicles includes a Mach 10 capability. The vehicles are recoverable and are used after refurbishment for subsequent tests. Flight data are recorded on on-board cameras and instrumented data are telemetered to ground-based stations.

The following summarizes the effort under each of the programs Goodyear Aerospace has conducted to develop expandable aerodynamic-type decelerators.

MACH 4 PROGRAM

Rocket boosted missile tests of the full-scale Ballute system were conducted during 1960 and 1961. These tests varied from a trajectory of Mach 2 at 120,000 feet to a desired Mach 3.5 of 155,000 feet. These trajectories were obtained by the use of two or three Nike boosters mounted in tandem, dependent on trajectory requirements. Each test missile contained five channels of telemetry for transmission of the snatch load, drag load, dynamic pressure, acceleration and Ballute internal pressure during flight, and gear for water impact and recovery after landing in the Gulf of Mexico. Four missiles were fired, three with a two-Nike booster set-up and one with a three-Nike booster setup; each achieved the desired Mach number and altitude. Upon recovery of these missiles, further data were obtained from an on-board movie camera. Generally, the measurements corroborated the data obtained during the wind tunnel and analytical phases.

Four free-fall tests were conducted at Holloman AFB near Alamogordo, N. M. The missile was ballasted to weigh 500 pounds and was carried to an altitude of approximately 98,000 feet by a 1,000,000-cubic foot, 1 mil Mylar balloon. The balloon was launched by standard procedures and rose at approximately 800 to 1000 fpm. At the desired altitude, the missile was severed by an explosive and allowed to fall free to about 88,000 feet, where its velocity was approximately 600 fps. At this point, a timer sequenced a second pyrotechnic charge that separated the segmented canister from the aft portion of the missile, deploying the Ballute. One-half second later, an internal pressure bottle was fired, inflating the Ballute to its full 9-foot diameter. The Ballute deployed on its 15-foot suspension line, as recorded by an on-board camera and by optical coverage from three ground stations at the test site. The total drag, Ballute internal pressure, acceleration, and dynamic pressure from the missile were telemetered and recorded. At approximately 50,000 feet, the parachute was deployed and the missile floated down for a spike impact in the desert area. The purpose of these free-fall tests was to further check the system functionally and gather data on the subsonic and transonic speed regimes.

MACH 10 PROGRAM

Upon completion of the Mach 4 phase of the program, more advanced work was initiated covering the development and analysis of a system capable of operating in the Mach 10 flight regimes. The program included a comprehensive study of various materials capable of withstanding the 1500 F temperature expected to be encountered in the flight profile, and investigation of various problems inherent in the aerodynamic stability and inflation of a device traveling at hypersonic speeds. Here again, extensive analytical work was conducted and followed up with wind tunnel tests at the Langley 4 X 4-foot Unitary Tunnel Mach 4.6 facility. Dacron-neoprene balloons of various sizes and configurations were tested first. The shapes to be considered now were essentially the basic sphere, cones of various angles, hemispheres, and other devices that would maintain inflation throughout the trajectory and down to sea-level altitude.

PERFORMANCE AND DESIGN STUDIES

Of the many wind tunnel models tested, the closed pressure-vessel type spheres, cones and hemispheres performed in a very stable manner behind a payload. The aerodynamic drag agreed well with the data of previous rigid model tests. A new type of Ballute system was evolved so that the recovery system could be limited to one stage, with the first stage decelerator also serving as the final stage decelerator at sea level.

This required the Ballute to increase its inflation pressure during the trajectory to offset the increased ambient pressure and increased dynamic pressure dictated by the velocity and altitude. In this case, the obvious solution was to utilize the same techniques as used on the parachute; that is, to use ram air (dynamic pressure) to inflate the body to its initial shape and to maintain its inflation down to sea level. Subsonic tests indicated that an opening somewhere in the vicinity of 2 or 3 percent of the total body diameter, fixed in place by a metal or fiberglass ring with a spider in the center of this ring attaching it to the single-line suspension mechanism, would be adequate to inflate the Ballute properly and maintain its inflation under a wide range of dynamic pressures.

However, as speeds surpassed Mach 2, problems were encountered. A phenomena seemed to exist in which fabric flutter and fluctuation in load level caused structural failure due to shock swallowing and variations in mass flow. This created a rather delicate problem in the stress analysis.

Various ram-air flexible and rigid models were constructed and equipped with transducers sensitive to rapid fluctuations of internal pressure. It was found that if a screen or reed-valve type inlet was installed in the inlet portion of the Ballute, these pulsations could be reduced or eliminated. Utilizing screens or reed valves, depending upon the application, the Ballutes were

successfully inflated and performed properly during the wind tunnel tests, which were designed to simulate as closely as practical the conditions established by the IBM computer trajectory study. The studies included plots of velocity versus altitude, Mach number versus altitude, g's versus altitude, time versus altitude, dynamic pressure versus altitude, and total pressure versus altitude, for various ballistic coefficients such as 1, 10, 50, and 100 and at four different initial conditions: Mach 4 at 70,000 feet, Mach 4 at 120,000 feet, Mach 10 at 120,000 feet and Mach 10 at 200,000 feet. In addition, the surface temperature distributions of spheres and conical bodies with and without forebody interference were analyzed.

Body temperature distribution was plotted as a function of altitude, velocity and Mach number for various initial conditions. In addition, optimum structural requirements for the Ballute were established. The optimum configuration became an 80 degree conical-shaped Ballute with an isotensoid, or in some cases, a tucked, aft portion of the Ballute. In an isotensoid surface, the fabric has uniform stress in all directions and the meridian cables have constant tension throughout their lengths. This allows a minimum weight design that takes the same shape when inflated as the shape to which it was tailored, and maintains this shape after inflation. It is capable of withstanding the high dynamic loads in opening. Essentially, an improved type of high-speed parachute was developed that retained the parachute weight and packaging features and yet overcame parachute shortcomings with respect to supersonic stability and aerodynamic heating resistance.

The payload was suspended from the fixed inlet in the Ballute by a single line connected to a number of cables that went around the Ballute to the middle of its aft portion. At this point, an internal cable joined the multiple cables and carried a portion of the load to the forward point of the inlet to distribute the load evenly through the Ballute fabric.

One other consideration was investigated: the inflation position of the Ballute at very high altitudes where the dynamic pressures would be extremely low. This problem was solved by installing a simple lightweight bladder, made of plastic or very thin rubber, inside the Ballute. Inflated with alcohol or other vaporizing liquid, the bladder erected the Ballute and positioned it so that it was deployed aft of the payload body in a wake field that provided adequate ram-air pressure for additional inflation. As the ram-air pressure became stronger than that in the internal pressurizing bladder, the bladder was compressed in the rear portion of the Ballute.

Of the other configurations tested at Mach 4.6, it was found that the hemispherical balloon and the 75- and 80-degree conical balloons were very stable. The 90-degree cone exhibited some instability and a very slight resultant increase in drag coefficient that precluded its further consideration. Essentially, the most promising results were in the area of either a positively inflated sphere or an 80-degree apex angle, ram-air inflated, conically-shaped balloon. The ram-air inflated balloon obviously had advantages over the other types due to its minimum inflation mechanism requirements, and is recommended for most applications.

HIGH-TEMPERATURE MATERIALS DEVELOPMENT

To further the deceleration studies into the Mach 10 regime, a development program was initiated to study high-temperature materials. Previous Goodyear Aerospace work in the field had narrowed the study into the area of woven wire meshes coated with a high-temperature, silicone-ceramic elastomer. These developments were explored further for the Ballute application, and a modification of the basic cloths was made. Numerous fine threads of wires (1.6 mil of René 41, a nickel-chromium alloy used primarily in turbine blades of jet engines) were twisted in groups of 7 strands into a yarn, then woven into a cloth with approximately 160 ends of wire or cable per linear inch. This multiple stranded or cabled cloth was then coated with a Goodyear Tire and Rubber Company development, a high-temperature silicone-ceramic elastomer, CS105.

This coating not only provides a sealant but also a capability to withstand approximately 8 psi differential pressure within the Ballute without gassing or excessive leakage of the woven wire mesh. Fabrication techniques were investigated and it was found that resistance spot welding in two staggered rows of approximately 8 welds per linear inch was optimum for about a 90 percent seam efficiency. Various mockup units and models were built of this material for structural testing. Ultimately, Goodyear Aerospace constructed five René 41 balloon models, 10 inches in diameter.

HYPERSONIC WIND TUNNEL TESTS

The Mach 10 tests were conducted at Arnold Engineering Development Center in the 50-inch Von Karmen tunnel. A unique feature of this tunnel is the model installation chamber below the test section, which allows the entire pitch mechanisms (sting and model) to be raised into and lowered out of the tunnel. When the model is in the retractive position, the fairing doors and safety doors can be closed, allowing entrance to the tank for model changes while the tunnel is running. When the model is in the test section, only the fairing doors are closed, leaving the tank at static pressure. Stagnation pressures of up to approximately 2000 psia are supplied.

The external geometry for all the Ballute models was basically similar, including the solid (heat transfer and pressure rigged) models and the flexible René 41 units. The heat transfer models were fabricated from 321 stainless steel and formed by the spinning process; the skin thickness varied from 0.5 to 0.064 inch. Nineteen thermocouples were spot welded on the interior surface and two were mounted inside the model to measure the internal ambient temperature. The pressure model also was fabricated from 321 stainless steel and was instrumented with 17 static orifices on the surface, and one internal orifice for measuring the internal pressure. Flexible drag models were fabricated from René 41 cloth. The seams of the models were joined by spot welding and were impregnated with a high-temperature silicone polymer to decrease the porosity. For added strength on the flexible models, eight longitudinal cables were connected from the nozzle to the base plate.

The inlet configuration used in the Mach 10 program was tested on the pressure model with screens covering the inlet slots; with reed valves, screen removed; and with open slots. In addition to the basic inlet, two types of inflating tubes were used with the flexible models. They were located either at the forward end or at the equator of the Ballute. The models were supported by a strut assembly. The strut leading edges of the vertical sections of the sides and the top of the horizontal section were water cooled. The flexible drag models were connected to a 3/32-inch diameter René cable routed around pulley wheels inside the strut. The cable could be extended or retracted by means of an hydraulic cylinder mounted in the bottom portion of the strut. The pressure and heat transfer models were supported with a 1.75-inch diameter sting. With a collet-type sting replacing the hemispherical cover, the sting lengths of the heat transfer and pressure models could be adjusted as desired.

The instrumentation of the models included a pressure data system that consisted basically of nine channels, each of which was time shared between 11 model orifices by means of a 12-position pressure switching valve. The total capability was 99 model measurements with the first position of each pressure switching valve being used for transducer calibration. A conventional short-range divergent ray sparked shadowgraph system was used to record selective flow patterns about the model. A strain gage type link was used for drag measurements on the flexible models. The drag link was incorporated and instrumented with two strain gage bridges and mounted on the hydraulic cylinder piston rod.

The tests were detailed and generally conducted at 0 degree angle of attack through a Reynolds number range from 0.36×10^6 to 1.8×10^6 per foot. The pressure data were obtained at various distances from the forebody and heat transfer data were obtained at an L/D of 18 (length of riser line to diameter of the forebody). The forebody and strut assembly were removed and additional heat transfer pressure data obtained over this same Reynolds range. The rigid and flexible models were installed and injected into the tunnel at the minimum desired L/D. The cable then was extended and drag measurements obtained over the desired L/D range. Some of the flexible models were preinflated and other models were inflated with the ram-air device. To keep the models relatively stable during the injection, a restraining cable was loosely connected from the model base to the hemispherical tie-down cover to the aft portion of the sting.

During the testing of these Mach 10 devices, it was found that the forebody wake did not tend to collapse or recover at any distance aft of the payload that was capable of being tested within the tunnel. In an attempt to collapse this wake, a number of experiments were attempted without success, such as welding steel balls or little scoops around the aft portion of the forebody in an effort to force and collapse the wake. However, neither of these did anything to collapse the long-wake core. This wake did not affect the positively inflated models (preinflated 20 psia models), as the stability of all the models was excellent. However, in order to inflate the ram-air models, a different method of ram-air inflation was required, due to the fact that this core existed over and outside the ram-air inlet diameter.

In order to overcome this problem, a spider network of small steel tubes was located around the periphery of the equator of the Ballute and the resultant ram-air pressure was fed down into a manifold on the aft portion of the Ballute, fed in turn through a screened inlet to the Ballute itself. This method of extended ram-air inlets worked very well. The test was conducted with the Ballute coupled closely to the forebody and also at various L/D ratios out to 18. In all cases, the inflation was positive, the Ballute fully inflated and stable. The drag coefficient for an 80-degree Ballute at this Mach 10 regime for the Reynolds numbers as described essentially runs approximately 0.7 to 0.8 as calculated on the base area of the model.

The latest Ballute configuration is an isotensoid design in which there are external meridian cables only. There is no internal cable. The ram-air inlets are located at the equator of the Ballute, because in this position they will be as far out of the forebody wake as possible, and will thus operate most efficiently in this position. The particular inlet shape was chosen to obtain a nominal orifice coefficient considering the boundary layer thickness around the Ballute.

PRESENT ADDPEP EFFORT

State-of-the-art advancement of aerodynamic deployable decelerators (i.e. various parachutes and balloon-type drag devices) requires two definite steps under this program: (1) establish the validity of available analytical and wind tunnel data through a free-flight test program using test missiles capable of achieving test point conditions and obtaining necessary performance data; and (2) if the available data are invalid, generate new analytical data to support wind tunnel tests.

Under the ADDPEP program Goodyear Aerospace will also establish and provide service to support a free-flight test program. It will fabricate and instrument test vehicles, provide decelerator test items, conduct exploratory tests where necessary, analyze and correlate technical data. The company will also establish or modify existing series criteria and performance prediction approaches and document program results that will include the redesigns or new designs required to achieve the goals previously mentioned.

Free-flight testing will be conducted at the Air Force Proving Ground Center Gulf Test Range, Eglin, Florida, for the rocket boosted test, and at Kirtland Air Force Base, New Mexico, for the aircraft drop tests with the heavy 2100-pound vehicle.

The small supersonic parachutes to be investigated are of the hyperflo variety (special ribbon type parachutes) and are constructed of HT-1 material to withstand the aerodynamic heating encountered at Mach numbers of 2 to 3. It is anticipated, that combinations of coated HT-1 and/or metal fabric will be required to withstand the temperature ranges above 700 F in the Mach 3.5 and upward regimes. This area is under investigation at the present time, including the fabrication technique and stress analysis associated with this specialized parachute construction.

Since analytical treatment for Ballutes has progressed to a degree approaching that of rigid structures, a much reduced program is planned that will be centered around improving the design and analytical techniques. Flights are planned for Mach numbers from 2 to 4 at 80,000 to 100,000 feet, to vary with the design models. This in turn will lead for an extension of the performance capability of the Ballute to Mach 10 in the free-flight case.

Wind tunnel tests of high dynamic pressure parachute models were recently completed and are under evaluation and consideration for testing at a later date. This includes a capability of 4000 psf "q" with parachutes 16 to 24 feet in diameter. The presently planned full-scale flight tests of the large hi-q parachutes will utilize modifications to the present USAF 16-foot hi-q parachute designs. This test treatment will afford an evaluation of missile data recording systems as well as confirming expected performance predictions for the large hi-q parachutes.

Ultimately, the purpose of the ADDPEP program is to come up with the documentation of recommended designs capable of fitting and fulfilling a multitude of the needs in the recovery area that heretofore have been unexplored. The ADDPEP effort will provide the space industry with vitally needed design and test data for advanced recovery systems.

APPLICATION OF THE BALLUTE DECELERATION SYSTEM TO THE GEMINI PROGRAM

The Gemini space capsule manned by two astronauts contains ejection seats that provide off-the-pad ejection capability in case of booster malfunction or any other malfunction upward from sea level to about 60,000 feet at Mach numbers of approximately 3, and also for the descent portion below 100,000 feet during the final recovery of the Gemini capsule in case of other recovery system failures.

The recovery of the astronauts in this case presents the problem of stabilization and deceleration of the man during the high-altitude, free-fall portion of his emergency escape sequence. Past experience and limited tests have shown that it is possible for a man, particularly an astronaut when rigged in a full pressure suit with the associated survival and egress equipment, to reach a point of instability during the free-fall from high-altitudes.

This instability can result in a coining and spinning motion which could exceed the human tolerance and injure the man fatally. The purpose of the Ballute system, which is attached directly to the Gemini astronaut, is to deploy immediately after astronaut separation from the ejection seat and inflate. It thus provides the stabilizing moment necessary to keep the man from achieving a coining or spinning position during his free fall until he reaches a lower altitude, where the Ballute will be released and the final recovery chute deployed at approximately 10,000 feet.

Among major problems Goodyear Aerospace is investigating for McDonnell Aircraft, capsule prime contractor on the Gemini program, is definition of the minimum Ballute size required to stabilize the ejected Gemini astronauts. This investigation must also determine the minimum riser line length between the Ballute and attachment to the men's backboards so as to preclude any possible tangling during the free-fall portion of the emergency recovery. The system must be of minimum weight and volume, and the Ballute system has had to be optimized to provide a weight of approximately two pounds, including all gear required in the Ballute, and a packaging volume of about 100 cubic inches. This effort requires a number of free-fall drops with various size Ballutes to define, at least in the subsonic regime, a minimum size that will provide adequate stabilization.

An 18-inch Ballute appears to do the job adequately. However, further analysis and testing in wind tunnels showed this size to be marginal during various attitudes of a man's arms and legs, so a rather unusual method of gimbaling and mounting a model man in a wind tunnel to allow freedom of movement in three planes was devised and a system for remote deployment of the Ballute during the spinning or tumbling motion of the model was developed.

It was then found that a riser line length of a minimum of 16 inches and a 36-inch diameter Ballute of equivalent size full scale would repeatedly pull the man out of a spin and stabilize. It was established as a criteria that in a true ejection case the man would not be allowed to spin up because the Ballute would be deployed prior to his achieving any such motion. Also it would be conservative and practical to have a Ballute capable of pulling a man out of a spin. These were the criteria observed in defining the performance requirement of the Ballute.

After a number of subsonic tests were conducted and many combinations of attachment points and riser line lengths tried, the tests were enlarged by the utilization of the 4 × 4-foot NASA-Langley supersonic tunnel. Here was used a similar three-degree-of-freedom type of gimbal set-up. It allowed the dummy to spin under supersonic flow conditions, and by remote deployment the Ballute pulled the dummy out of the spin and held it in the stabilized condition.

Upon conclusion of these tests a full scale system was designed and fabricated that was tested in the AEDC 16 by 16-foot supersonic tunnel. In this series of tests, a 6-foot dummy rigged to duplicate as closely as possible an ejected astronaut, was mounted in the tunnel in a gimbaling system similar to that used in subsonic and supersonic tests. The dummy included comprehensive instrumentation.

The purpose of these tests was to verify and to functionally test the actual prototype Ballutes that will be used in the qualification phase of the program. Much was learned during this program as to selection of the lightest weight and most packageable fabric, the best seaming technique, method of folding and packaging the inlets and the associated filming, and the method of deployment from the canister attached to the backboard behind the astronaut's left arm. However, one of the basic problems in the wind tunnel tests with a decelerator is that it is difficult to achieve the actual condition of free flight - the condition of a decaying high-dynamic pressure flow field. The constant "q"

condition in the tunnel creates unrealistic loads on the decelerator so the test decelerator must be overdesigned to withstand high dynamic pressure exposure over a period of time.

These tests established the qualifiable Ballute system and the next phase of the program is the development and qualification free-flight drop testing from B-66 drop-test type aircraft and C-130 aircraft, followed by live drops by test jumpers. This program will consist of a number of instrumented dummy drops from altitudes from 45,000 feet at similar dynamic pressures experienced during the upward trajectory of the Gemini capsule. The instrumented dummies will contain two motion picture cameras, one pointing upward and the other one pointing downward. The telemetry system on board the dummy will transmit information from accelerometers and rate gyros that will be used to augment stability analysis of the Ballute-astronaut combination.

It must be noted that the man-Ballute combination is a very complex aerodynamic body and defies very accurate aerodynamic calculation. Tests are required to verify assumptions in every phase of the program. It is significant that the application of an expandable structure for an application such as the Ballute in stabilizing a Gemini astronaut is just one of the many far-reaching capabilities that exist in the field of expandable structures.

In this manner the technology of fabrics, whether high temperature, high strength or lightweight, can be applied to provide an optimum system consistent with the practice required on space vehicles.

AN ELASTIC RECOVERY CONCEPT FOR EXPANDABLE SPACE STRUCTURES

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INTRODUCTION

A Program for the development and evaluation of the elastic recovery concept for expandable space structures was initiated by the Office of Advanced Research and Technology, NASA Headquarters, 18 March 1963. The objectives of this program are

1. To select optimum composite designs utilizing the elastic recovery principle for expandable space structures
2. To compare the elastic recovery concepts with other expandable concepts
3. To select optimum applications of the elastic recovery concept for space structures

The feasibility of this approach has been established. Design techniques are being developed, and applicable materials have been selected. Approaches to design and structural applications are discussed in the following text.

DEFINITION OF THE ELASTIC RECOVERY CONCEPT

The elastic recovery concept sandwiches a compressible core between two or more flexible facings. The materials used in the construction permit the structure to be alternately folded and compressed into an extremely small-volume container. Upon release from the container, the stored potential energy of the compacted material is sufficient to expand and rigidize the structure.

This method of expansion and rigidization has many advantages with respect to the others, some of which are listed below:

1. There is an extremely high expanded volume to packaged volume ratio; i.e., between 30:1 and 100:1.
2. This method offers an inherent expansion and rigidization mechanism with no auxiliary forces required such as gas pressurization, chemical reaction or mechanical devices.

3. The method is adaptable to multi-wall construction for ultimate meteorite and radiation protection.

4. The structure is extremely lightweight.

5. Construction is based on available state-of-the-art materials, which are compatible with space environments.

6. The approach is adaptable to standard manufacturing procedures.

7. The method offers reliable expansion, incorporating fail-safe principles.

The most efficient designs will utilize plastic films, elastomeric cores, and reinforced plastic laminates. The basic components of this design, illustrated in Figure 1, are the compressible core, inner pressure shell, meteoroid shield, and thermal protection.

A. Compressible Core

The core is the mechanism for both expansion and stabilization. There are a variety of cores applicable in this construction. Flexible foam is the most basic of cores and the most readily available. Figure 2 shows typical 1 lb/cu ft polyurethane foam compressed and expanded.

Another type of core that offers promise is flexible honeycomb. Figure 3 shows a typical sample of flexible honeycomb compressed and expanded. Many elastomeric materials can be used in this construction, such as neoprene, polyvinyl, reinforced elastomeric film, etc.

Elastomeric materials can be used to form other types of core such as the expanding wedges shown in Figure 4. These wedges can be expanded in two directions compared to the one direction of honeycomb.

Other applications of elastomeric materials as core structures are shown in Figures 5 and 6. A model of sandwich construction where elastomeric tubing is used as longitudinal supports is demonstrated in Figure 5. The use of elastomeric tubing helically wound between two facings is shown collapsed and expanded in Figure 6.

B. Inner Pressure Shell

The inner pressure shell must maintain flexibility, be capable of supporting pressure, and be impermeable for most space applications. The plastic film, normally used for pressure liners, must be reinforced in large structures to provide necessary strength. The plastic matrix used in this composite is selected for its degree of impermeability. Four of the most impermeable resin systems are

1. Polyvinylidine-chloride (Saran)

2. Polyethylene-terephthalate (Mylar)

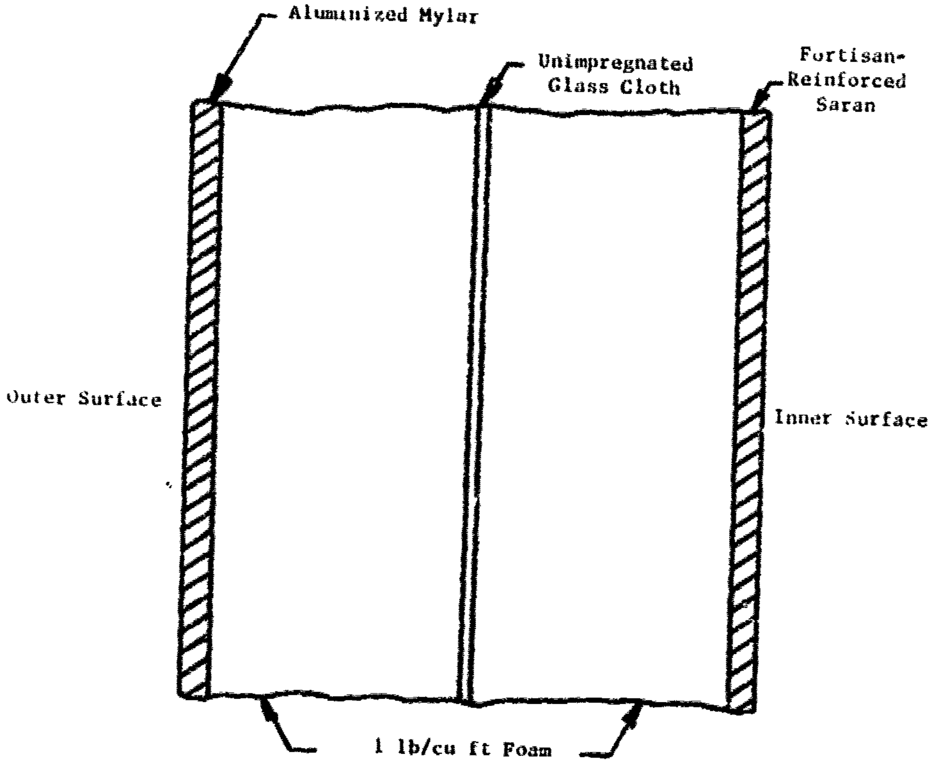
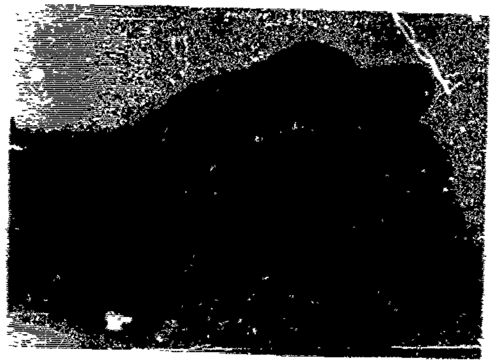


Figure 1. Typical Cross Section of an Elastic Recovery Design Concept - Narmco MEG-A-WALL

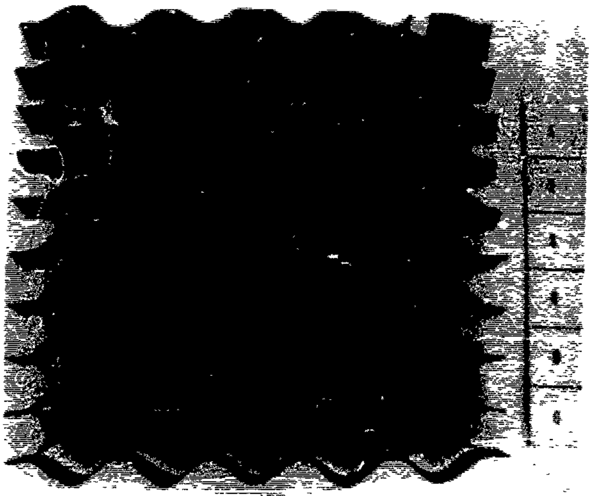


Compressed

Expanded

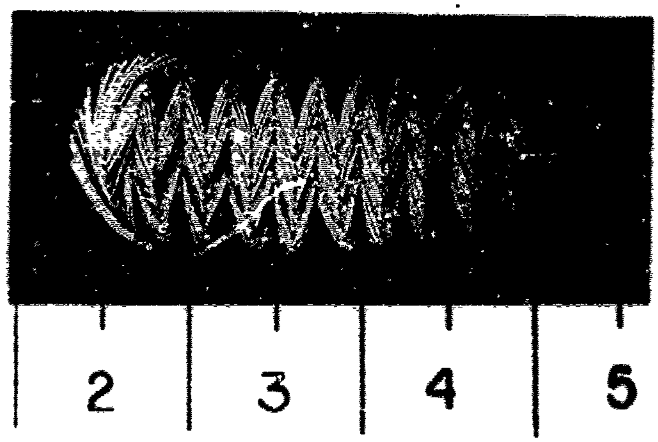
Figure 2. Foam Sample

Compressed

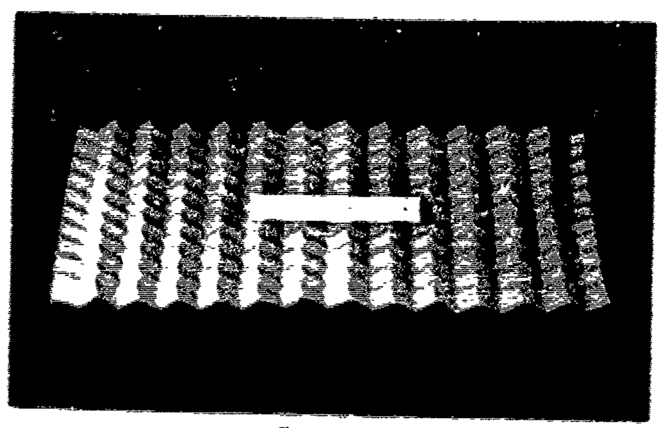


Expanded

Figure 3. Flexible Honeycomb

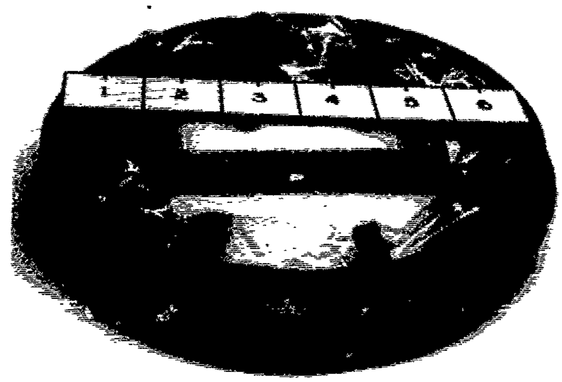


Compressed

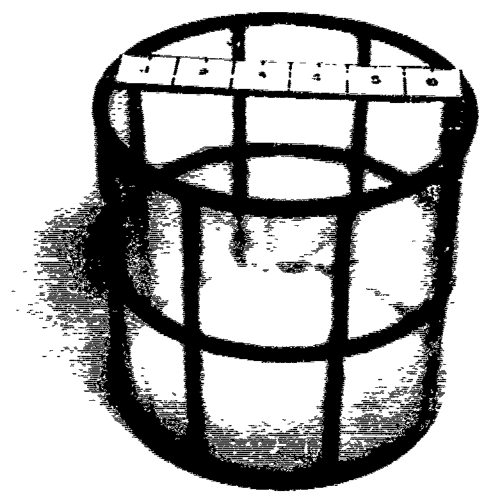


Expanded

Figure 4. Expanding Wedges Core Material



Collapsed



Expanded

Figure 5. Fixed Rings with Flexible Longitudinal Supports

Collapsed

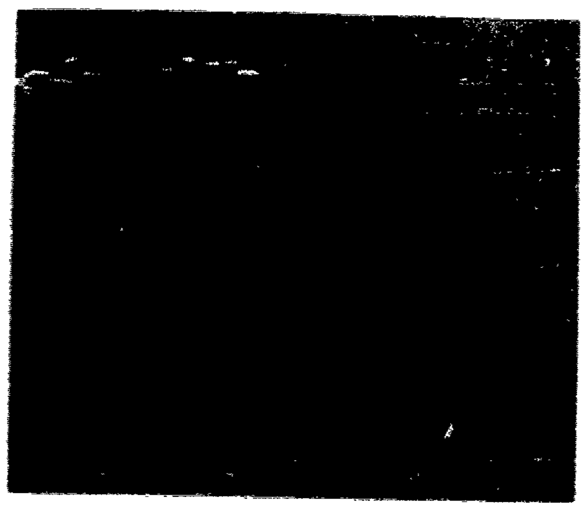


Figure 6. Fixed Rings With Helical Flexible Supports

3. Polychlorotri-fluoroethylene (Teflon)

4. Polyvinylchloride (PVC)

The time required for pressure to drop from 14 psi to 7 psi by gas leakage through different inner membrane materials of an 8-ft diameter, 15-ft long cylinder is shown in Figure 7. From this data, it is apparent that Saran is superior to the other materials.

Reinforcement can be provided by Fortisan, Dacron, or nylon fibers. Glass fibers are too brittle for use in most flexible designs where maximum packaging is required. Tests have demonstrated that Fortisan-reinforced Saran provides a very impermeable, tough, and strong-pressure membrane. A demonstration of the toughness of this material is presented in Figure 8. Even where the fibers have been ruptured, the Saran matrix still forms a tough, impermeable film. This laminate can be bonded to the flexible foam by conventional methods with a Saran resin system. The integrity of the bond is demonstrated in Figure 9.

C. Meteoroid Shield

Hypervelocity impacts can best be absorbed by a series of non-connected parallel plates where the outer surface acts as a bumper, shattering the particle. The space between the bumper and the next plate allows the shock wave to spread, reducing the impact force of the particles on the adjacent plate. If the impact force is reduced sufficiently, the intermediate plate will absorb the remaining energy. If the force is sufficient to penetrate the intermediate plate, the space between the intermediate plate and inner plate will allow the shock wave sufficient additional expansion to further reduce the force and, therefore, prevent penetration of the inner plate. This system is adaptable to rigid construction, but presents some difficulty in flexible construction.

Since a core is of an absolute necessity in this expandable concept, it must be one that will permit the penetrating shock wave to expand unrestricted, otherwise a collimation effect could be created, concentrating the force over a small area. The polyurethane foam would offer limited expansion resistance and would therefore perform similarly to a void. Also, the multiple cell walls of the foam provide some energy absorption for microparticles.

The expandable concept shown in Figure 1 has all of the design features required of a good bumper-type meteoroid shield. The thermal shield would act as a bumper, the foam analogous to a void, and the inner layer as the major energy absorber.

The energy-absorbing material must be impact resistant and flexible. Unimpregnated glass cloth is good in this application It is felt that the glass cloth on a flexible base of foam would be capable of not only absorbing energy by shattering, but by displacement and the transition of forces 90° to impact.

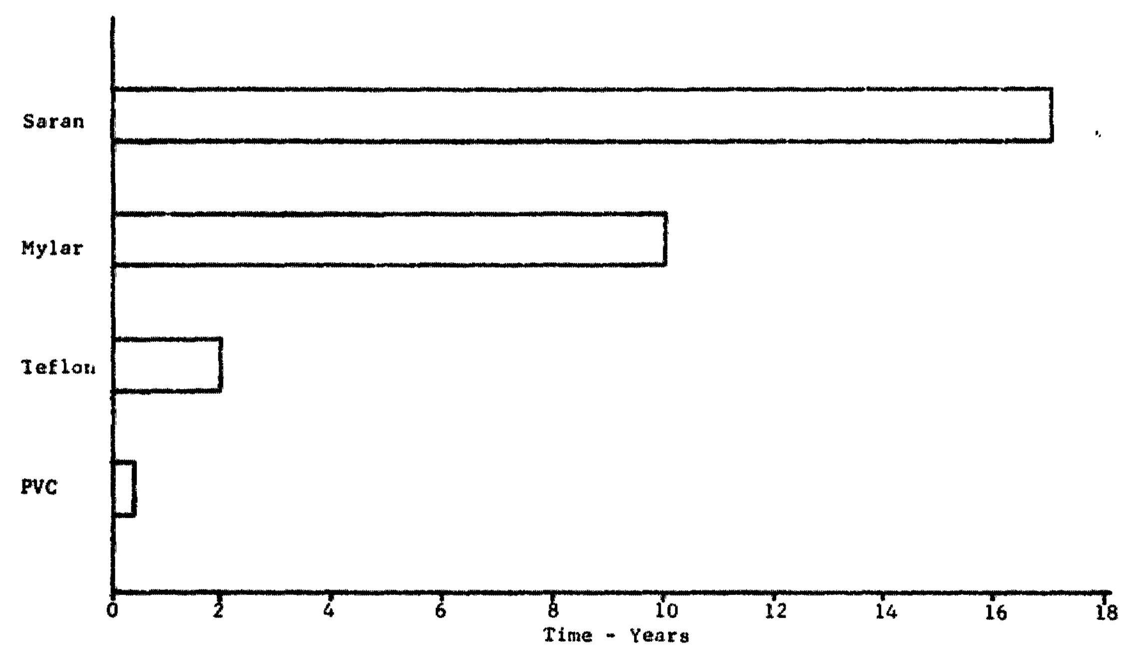


Figure 7. Time Required For Pressure to Drop From 14 psi to 7 psi by Gas Leakage Through Inner Pressure Membrane

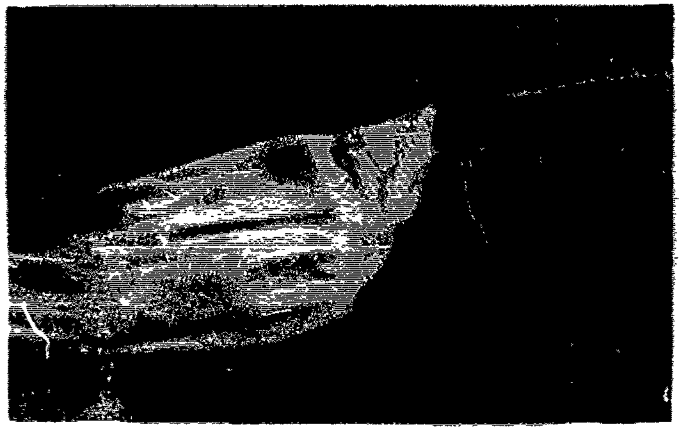


Figure 8. Toughness and Durability of Narmco 568

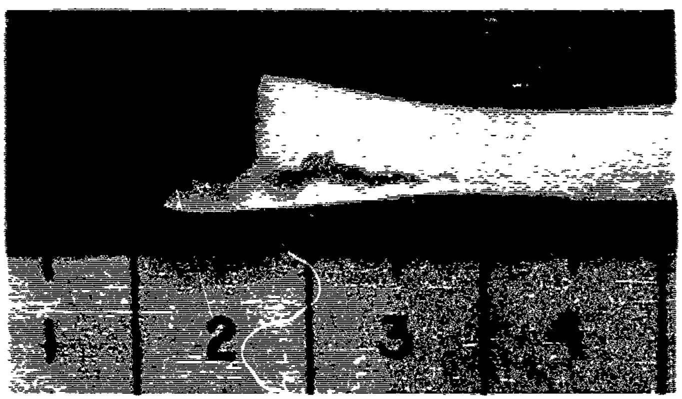


Figure 9. Bond Strength Between Shell and Foam

A variety of sandwich sections, similar to Figure 1, was fabricated and tested in the General Dynamics/Convair Division Shock Tube Facility. This facility is capable of impacting multiple 0.7-gm steel masses at 15,000 fps. These particles were approximately 0.013 in. in diameter. A variety of sandwich configurations was tested in this environment. The number of intermediate layers of impact absorbers varied from one to three at different levels throughout the sandwich. Glass, Dacron, and nylon cloths were all considered as candidates and were used as intermediate layers in these tests.

For the size particles and the impact velocities used, it was determined that the configuration shown in Figure 10 was the most efficient design for impact absorption. It consisted of a 0.01-in. thick aluminized Mylar outer shell, a 1-in. thick layer of 1 lb/cu ft polyurethane foam, 1 layer of unimpregnated 181 style glass cloth, another 1-in. thick layer of 1 lb/cu ft polyurethane foam, and a 0.05-in. layer of Fortisan-reinforced Saran cloth.

The particle penetration distribution is shown in Figure 10. Ninety-nine percent of the particles penetrated the layer of Mylar, eighty-eight percent penetrated the first layer of foam, five percent penetrated the glass cloth, and none reached the surface of the inner pressure layer. It is acknowledged that the particle velocities in these experiments are low compared to actual meteoroids. However, the capability of impact absorption is shown to exist.

D. Thermal Protection

Thermal controls must be applied to this construction if used for manned space structures. Since the major heating source is solar radiation, a high-emissivity coating is desired. One approach is to provide an aluminized outer surface. This can be accomplished by utilizing aluminized Mylar or Teflon as the external surface of the sandwich.

Experimental studies were conducted by applying quartz lamp radiation through a window in a vacuum chamber on an instrumented sandwich sample as shown in Figure 11. A thermocouple simulating a grey body was placed on the aluminized surface. Thermocouples were placed internally at the Mylar-foam interface and at the foam-Fortisan Saran interface.

The radiant heat was controlled so that the temperature recorded from the surface thermocouple was stabilized at 400°F. The inner thermocouple became stabilized approximately 2 minutes after the surface thermocouple was stabilized. The temperature behind the aluminized Mylar outer wall was stabilized at 90°F and the temperature at the foam-Fortisan Saran interface was stabilized at 70°F. While the spectral distribution of the radiation from the quartz lamp is centered at much longer wavelengths than that of the sun, this test gives a good preliminary demonstration of the effectiveness of the aluminized Mylar for passive thermal control.

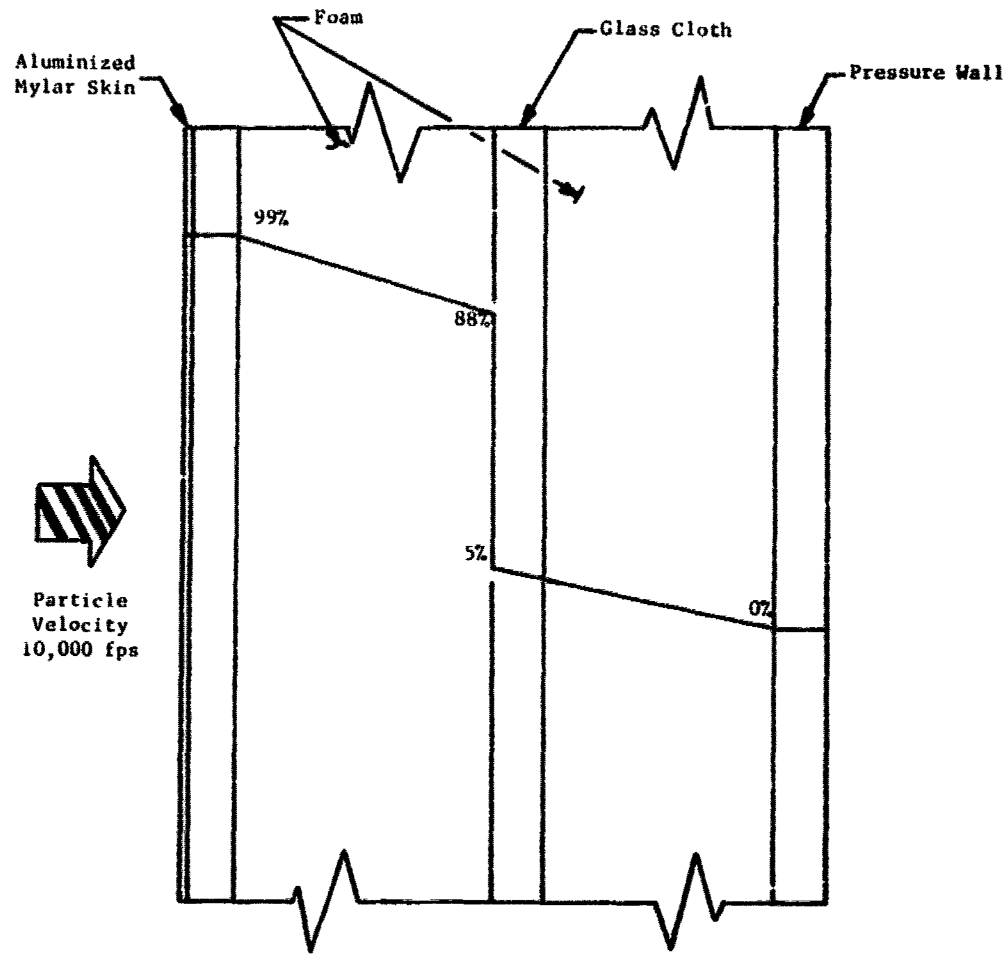


Figure 10. Meteoroid Distribution

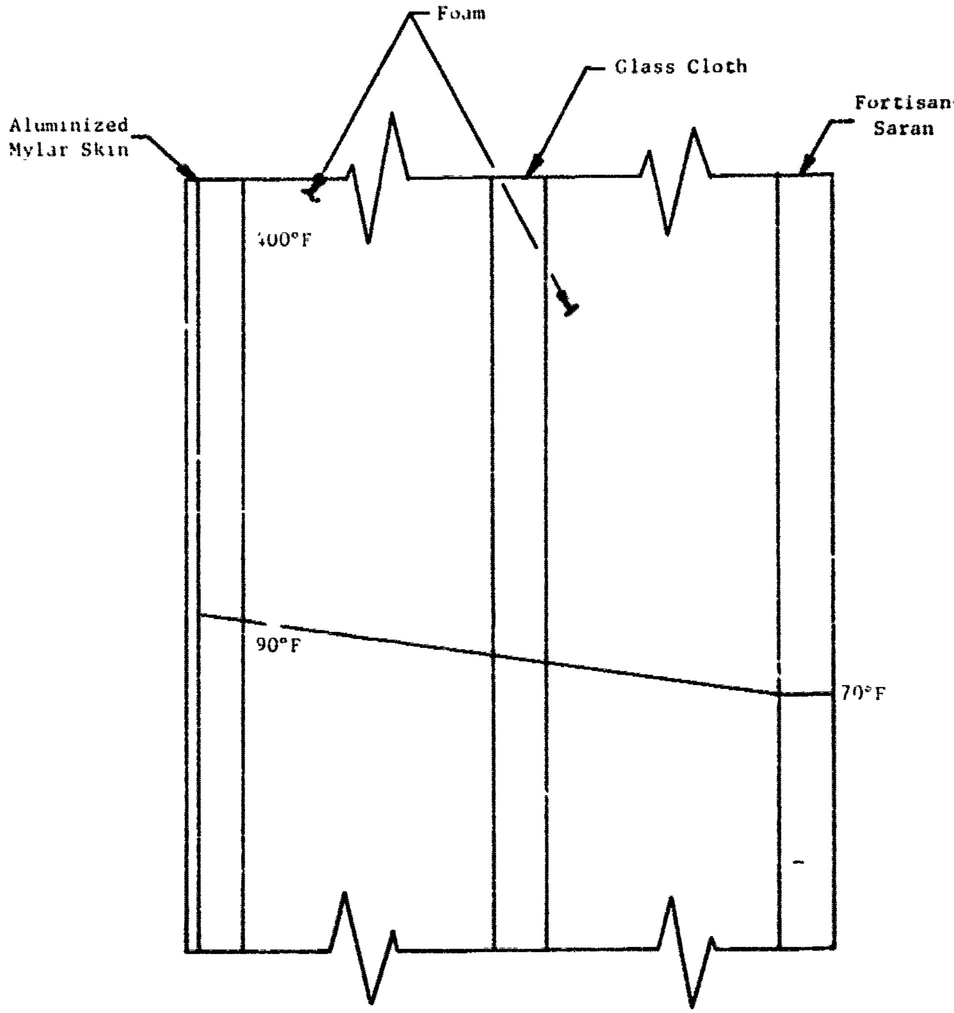


Figure 11. Temperature Profile Across MEG-A-WALL

APPLICATIONS OF THE ELASTIC RECOVERY CONCEPT

The utilization of a flexible-type expandable material for primary structures in an inhibited space station is doubtful for several reasons, which are as follows:

1. The ability of future boosters to support payloads up to 30 ft in diameter
2. The desire to have manned space systems assembled and checked out prior to launch
3. The psychological effect of a flexible material on an astronaut
4. The lack of reliability and service-life data

However, it is believed that expandable structures will be required and will find use where large volume structures will be needed. Some of these applications are presented in the following discussions.

A. Interconnector

Various space structures will require interconnecting chambers which would permit personnel to move freely through the system. One such system where an interconnector is applicable is the proposed manned orbiting research laboratory. The laboratory and booster would remain connected for zero gravity experiments. For artificial gravity experiments, the booster and laboratory would be disconnected and the interconnector allowed to expand, separating the two units as shown in Figure 12. The system would then be rotated, with the interconnector providing the structural tie. The booster could be used for logistic stores or space for experiments. Therefore, it would be desirable to provide an interconnector with an inhabitable atmosphere complete with micrometeoroid protection. A preliminary design approach for an elastic recovery concept is shown in Figure 12. Aluminized Mylar and foam will provide thermal protection. The foam and fiberglass will provide micrometeoroid protection, and an 0.02-in. thick pressure membrane of directional Fortisan-reinforced Saran will provide support for an internal atmosphere of 5.1 psi Rotational forces will be removed by lightweight Fortisan or Dacron fiber cables. A typical structure would weigh approximately 0.40 lb/sq ft of surface. A 10-ft diameter, 100-ft long interconnector could be packaged in a 10-ft diameter, 6-ft long container.

B. Space Hangar

Exploration and utilization of deep space will require space ports where vehicles launched from earth can be refitted and launched for continuance of their mission. It would be preferable if work on these vehicles could be conducted in an inhabitable atmosphere such as a space hangar, as illustrated in Figure 13. A typical approach to the construction of such a hangar is presented in this figure. Multiple layers of foam are provided for the additional elastic energy required for such a large structure. Additional micrometeoroid protection is provided by the multiple layers of glass cloth. Thermal protection is provided, as

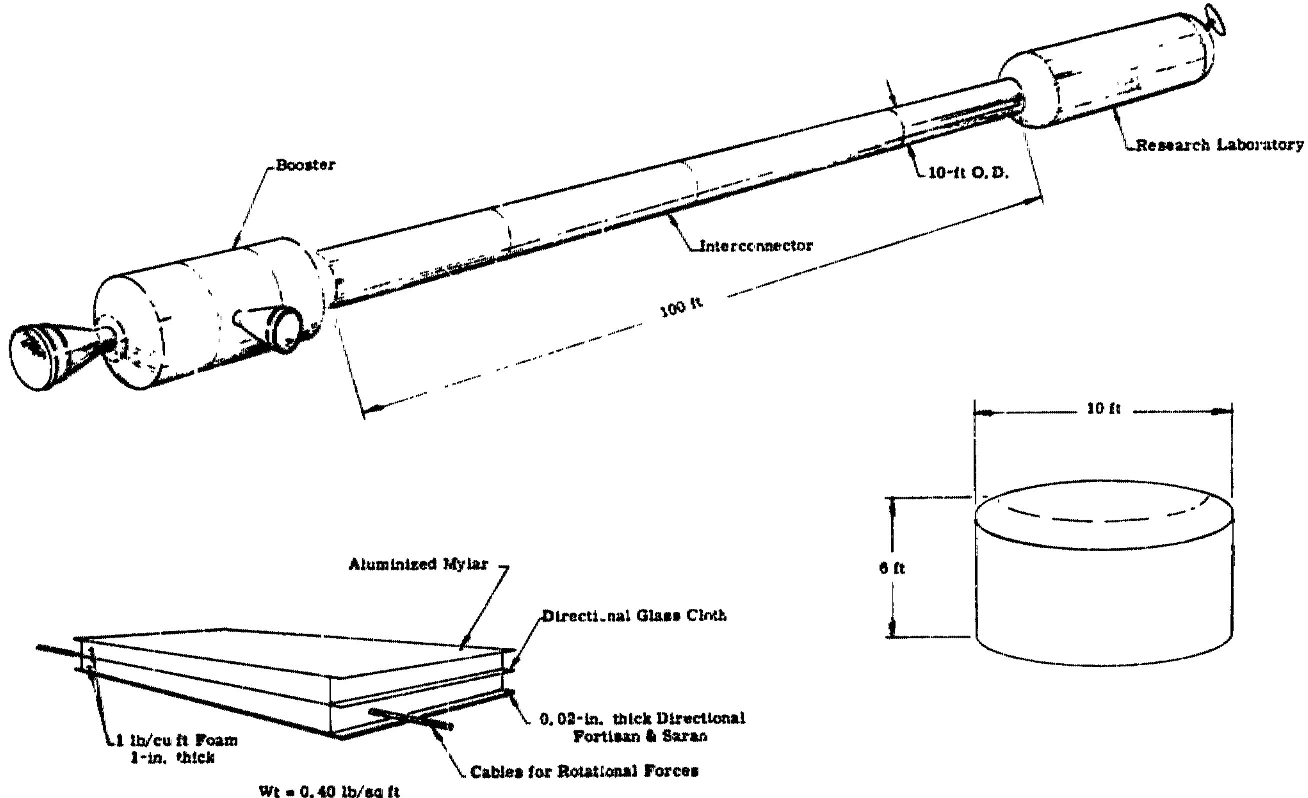


Figure 12. Interconnector

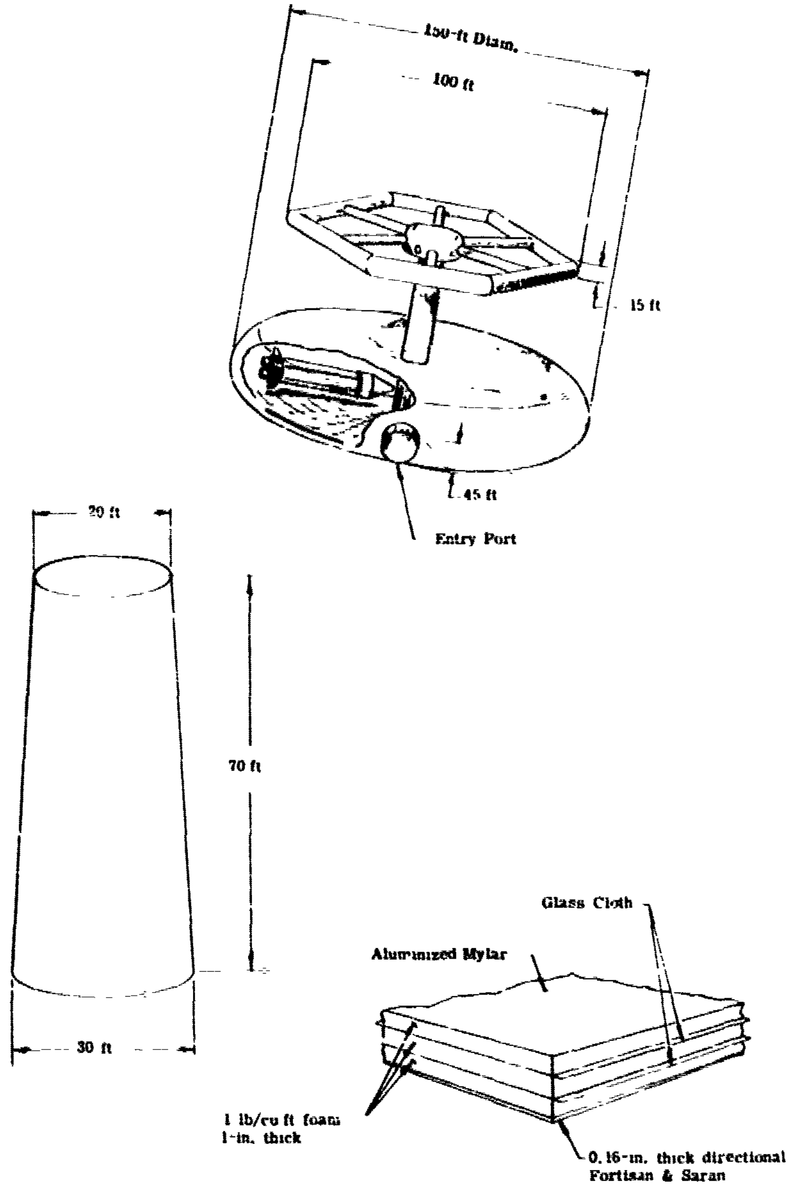


Figure 13. Space Hanger

previously discussed. A 5.1-psi internal atmosphere would require a 0.16-in. thick inner membrane. A space structure 150 ft in diameter and 75-ft high, as shown, would weigh approximately 1.70 lb/sq ft of surface area and could be packaged into a volume approximately 30 ft in diameter and 70-ft long.

C. Escape Capsule

To insure personnel safety in space flight, it will be necessary to develop escape capsules or "space life boats." One approach is that shown in Figure 14. An 8-ft long, 4-ft diameter capsule could be fabricated (as shown) which would provide thermal and micrometeoroid protection, an internal atmosphere of 5.1 psi, and would weigh only 0.25 lb/sq ft of surface area. This capsule could be packaged in a container 3 ft in diameter and 4-in. thick. The capsule could be equipped with transmitting equipment, logistic supplies, and even limited jet controls for maneuverability. The unit could be maintained in space until a rescue rendezvous could be initiated. This would be preferable to proposed concepts in which a man is foamed in a space package, and reentered into the earth's atmosphere. Also, this concept could be used for deep space probes.

D. Cryogenic Storage Tank

Logistic requirements for space programs will require that cryogenic fuels be stored at intervals throughout space mission trajectories. It would be desirable to provide large, singular containers which would keep the tank surface area to enclosed volume to a minimum. Permissible fuel losses are 0.32% per day for liquid hydrogen, and 0.08% per day for liquid oxygen. The wall construction shown in Figure 15 will limit the heat flow rate through the wall to less than 15 Btu/hr-ft^2. This will limit fuel losses to less than the minimum permissible rates for the 300-ft long, 60-ft diameter tank shown here.

This structure incorporates micrometeoroid protection and for a wall weight of 3.2 lb/sq ft permits internal pressures up to 30 psi. This structure could be packaged in a container 30 ft in diameter and 40-ft long for transportation from earth.

E. Antennas

Communication systems in space travel will require large antennas. One construction approach is the utilization of the helical-wound elastomeric tube system shown in Figure 16. The tubular system would provide erection and stabilization. The antenna surface would be of some metallic membrane such as aluminized Mylar, metal screens, or other reflective surfaces. A 300-ft diameter antenna dish could be constructed for a weight of 0.15 lb/sq ft. This unit could be packaged in a container 20 ft in diameter and 1-ft thick for transportation into space.

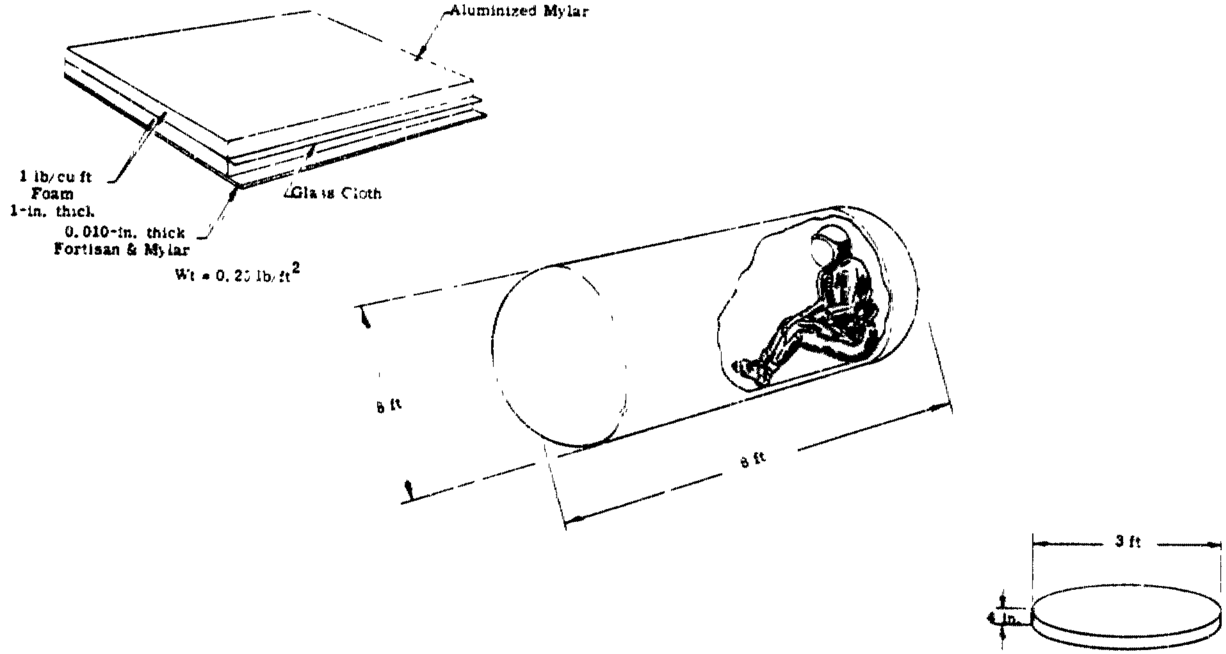


Figure 14. Escape Capsule

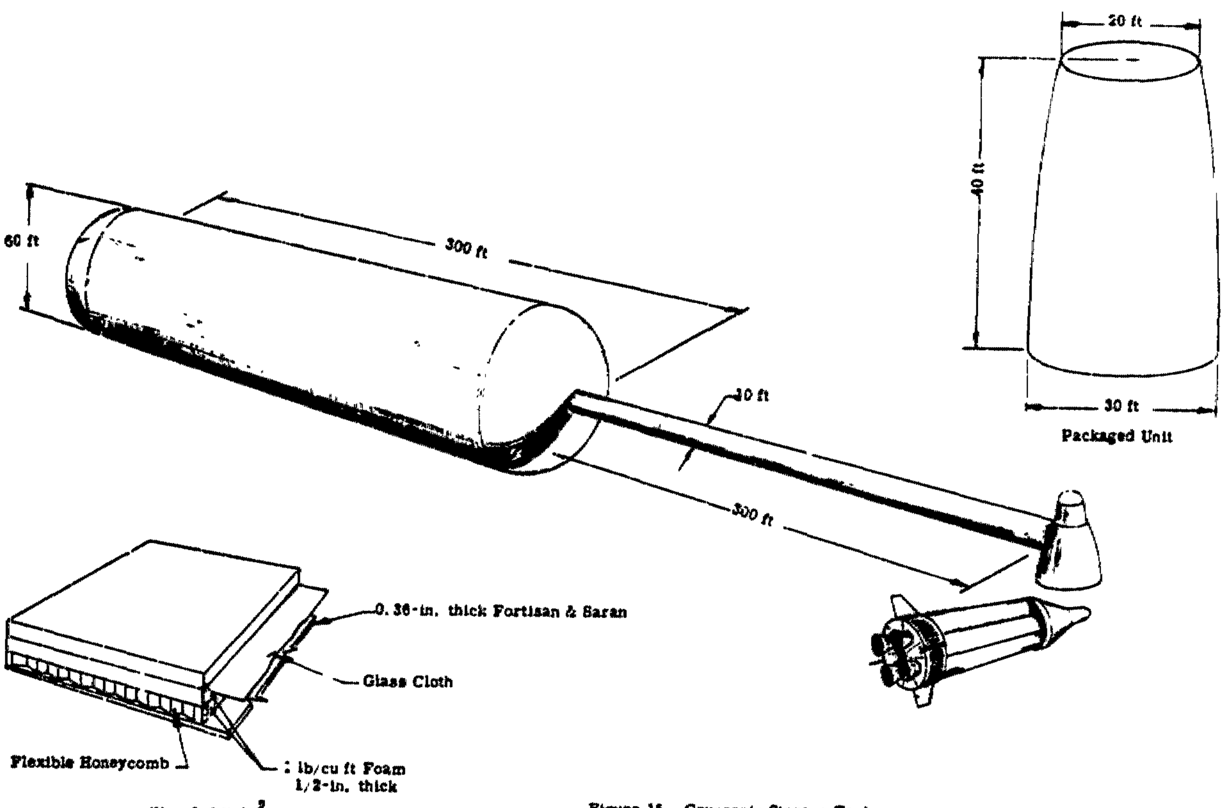


Figure 15. Cryogenic Storage Tank

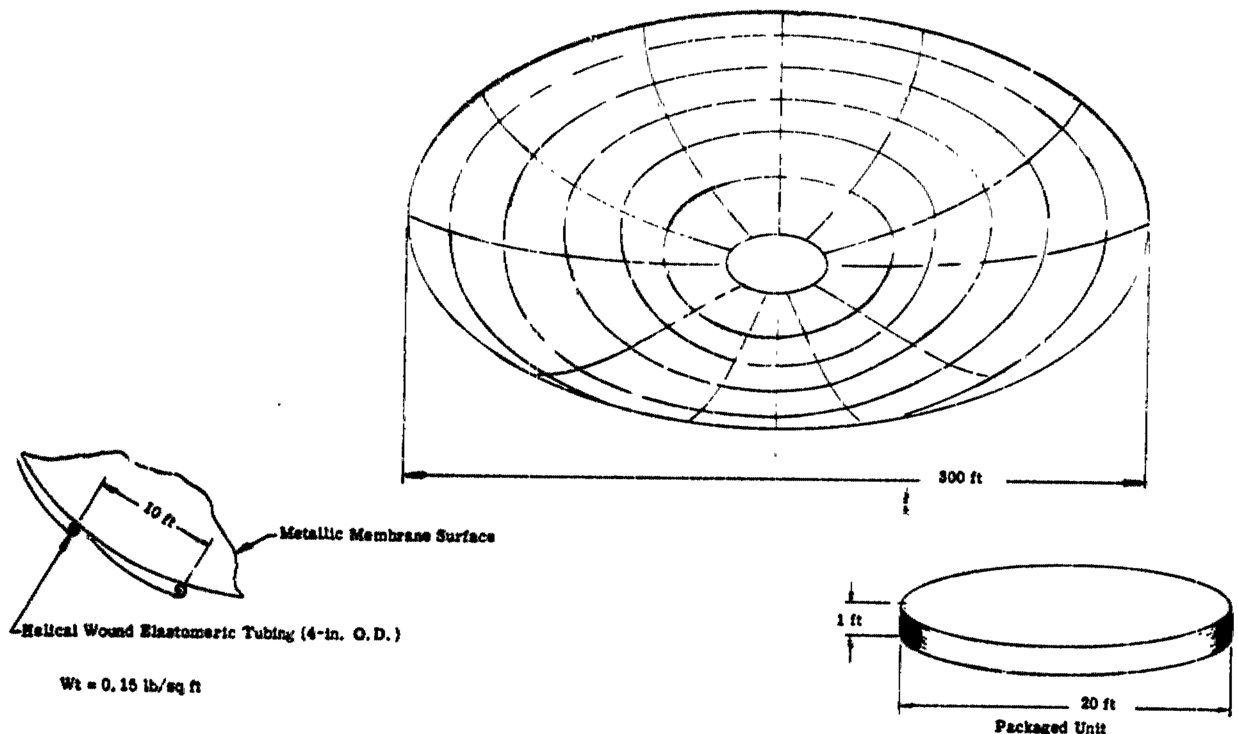


Figure 16. Antennas

F. Lunar Shelters

Exploration and exploitation of other planetary bodies will require the utilization of shelters capable of providing an inhabitable atmosphere. A preliminary concept for a lunar shelter utilizing the elastic recovery concept is shown in Figure 17. Multiple layers of foam and glass cloth are provided for particle penetration and thermal protection. Elastomeric honeycomb is provided for additional stiffness against planetary gravity. A structure such as the one shown could be constructed with a wall weight of 0.90 lb/sq ft. The packaged volume of an expanded 200-ft long, 40-ft wide, and 20-ft high structure would be that of a 10-ft diameter, 5-ft thick cylinder. This structure would not only be capable of simple space transportation but could also be collapsed and erected on the planetary surface as desired.

G. Flexible Mats

Transportation on planetary surfaces may create problems, especially where thick layers of dust are present. In order to move vehicles across the surface or to erect structures on the surface, a firm foundation may be required. This could be accomplished by the use of a mat such as the one shown in Figure 18.

The flexible honeycomb sandwiched by 0.10-in. thick facings of Fortisan-reinforced Saran could support 400 tons/sq ft for a structure weight of 2.1 lb/sq ft. This structure could be rolled up and collapsed into a very small volume, transported about a planetary surface, and used where required.

It is believed that in no other construction concept can an equivalent volume to packaged volume ratio be achieved with fail-safe erection, stabilization, and protection. All of these advantages are inherent in the construction with no auxiliary devices required.

The above applications are presented as examples where the elastic recovery concept can be used. Multiple other applications are visualized.

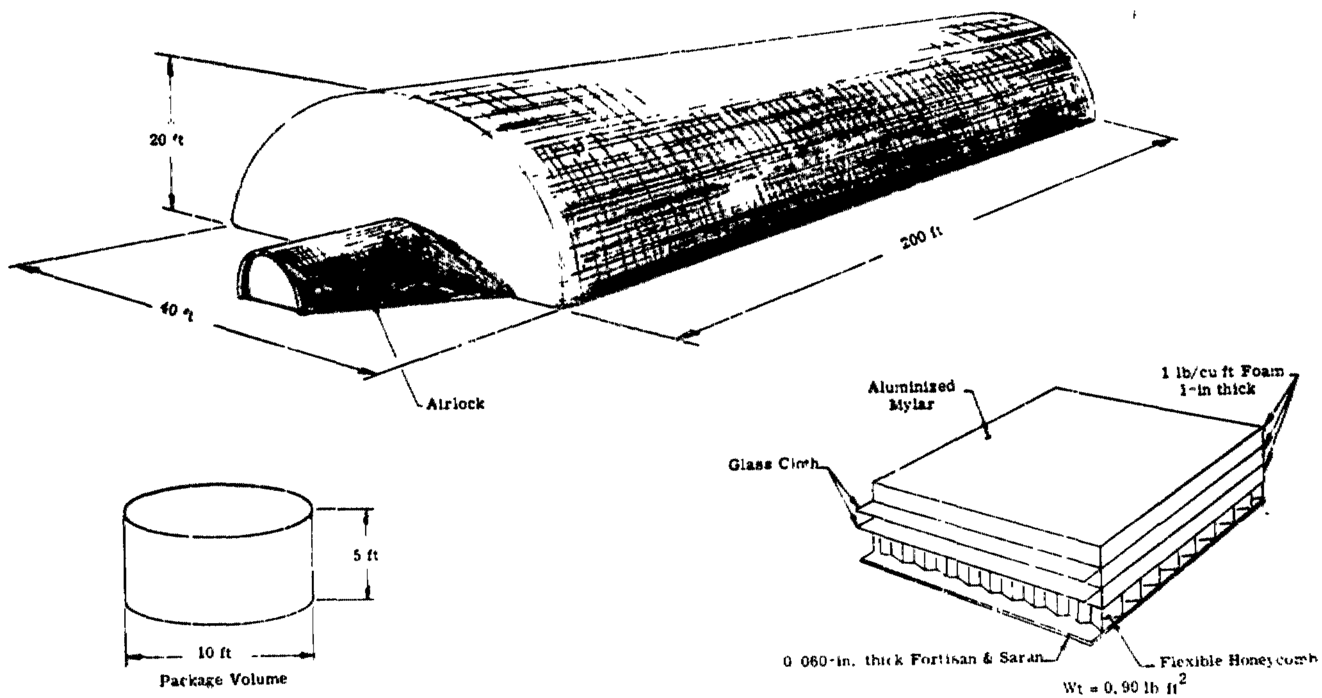


Figure 17. Lunar Shelter

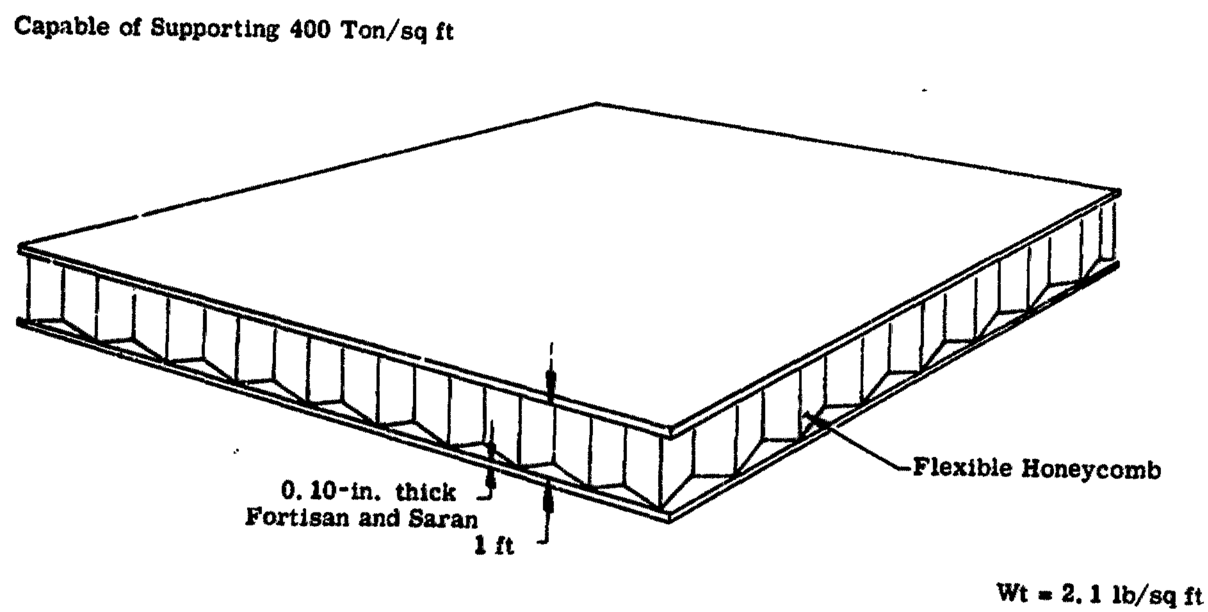


Figure 18. Flexible Mats for Soft Terrain

VARIABLE GEOMETRY SPACE STRUCTURES

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Northrop Space Laboratories

INTRODUCTION

Man, in his conquest and exploration of space, will require the use of structures to house, protect, and deploy various systems and vehicles. Each of these systems will employ many different structures to perform the multitude of functions necessary to satisfy the mission requirements.

A number of concepts have been proposed which fit into the general category of expandable structures. The structures can be further categorized into (1) inflatables or (2) rigid systems (Ref. 1). The inflatable is a balloon type which is intentionally pressurized to expand to many times its initial size. The rigid expandable structure is made up of rigid components compacted into a small package which can be rearranged to provide a greater surface area or enclosed volume. Two of the more familiar systems utilizing this concept are the solar panel array that deploys to an enlarged surface and the telescoping cylinder that provides an enlarged contained volume.

A system used in the space environment requires special design features. For example, protection from damage by micrometeoroids (Ref. 2) is frequently required to protect against puncture and deflation of a pressurized system. Even in instances where deflation is not a problem, it is generally necessary to provide some micrometeoroid protection for equipment on board the spacecraft.

Another special requirement resulting from the space environment is the need to protect the spacecraft payload from radiation damage by solar flares (Refs. 3 and 4). This may be satisfied by the addition of shielding taking the form of solid, rigid material.

Therefore, it appears that the space structure which houses subsystems must be rigidized and have a finite thickness generally governed by micrometeoroid or solar radiation protection requirements. In the event that an inflatable system is used, it must be capable of rigidization to be effective in meeting these requirements of the space environment.

There are spacecraft which require structure for no other reason than to position or locate components. Such is the case of an arm which deploys a solar panel: another, the antenna. These structures do not require pressure containment, but basically provide a means of obtaining orientation or deployment. The applied loads on these components will be very small. Nevertheless, the requirement for compact launch volume remains, and the structures must be deployed in space.

The study reported here concentrated on the variable geometry (VG) concepts which introduce to the structural engineer a new set of arrays that can be developed to form a broad spectrum of structural shapes for application in space, as well as -- perhaps -- for terrestrial use.

VG FRAME STRUCTURES

The basic structural elements of the VG frame structure system are the arch and base ring. By attaching arches around the base ring and providing the means of hinging and actuation, the arches can be rotated and made to assume the desired configuration. The arches lie essentially in one plane in the compressed state; in the deployed state the arches are rotated to form a three-dimensional structural framework.

SINGLE-STATE SYSTEMS

Two of the VG frame systems are shown in Figs. 1 and 2. These show the single-stage systems in a compressed and a completely deployed state.

It is possible to identify and evaluate a number of parameters associated with this system. The following discussion emphasizes the single-stage framework since it is a basic unit in the VG structural system. Some of the parameters of influence are as follows:

1. Variation of arch shapes
2. Out-of-the-plane arch curvature
3. Degree of interlacing
4. Manner of folding
5. Number of arches disposed about the base ring
6. Geometric variations of the base ring

Variation in Arch Shape

Fig. 1 shows the triangular arch system, and Fig. 2 shows the semicircular arch framework. Other shapes like the trapezoid, parabola, ellipse or optimized geometries for specialized applications may also be used. The trapezoidal arch system appears to be the most attractive for general usage since it can provide an essentially constant cross-section along the axial length of the deployed structure.

Out-of-the-Plane Arch Curvature

The arches can be made such that they have curvature out of their plane. This results in a framework, in a deployed state, which would possess curvature and could have application in those cases where a dome-like framework is desired. On the other hand, the compressed package volume would be larger than the planar frame network.

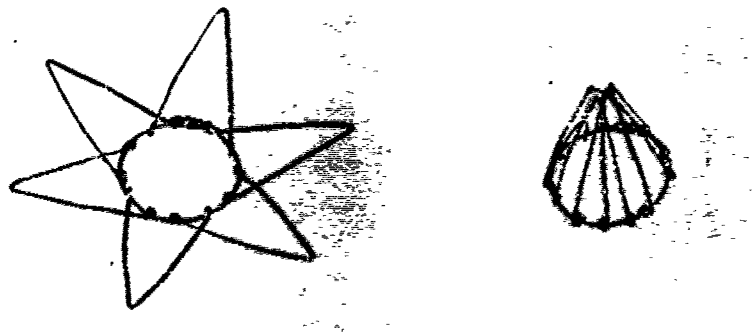


FIGURE 1 TRIANGULAR ARCH SINGLE STAGE VG STRUCTURE

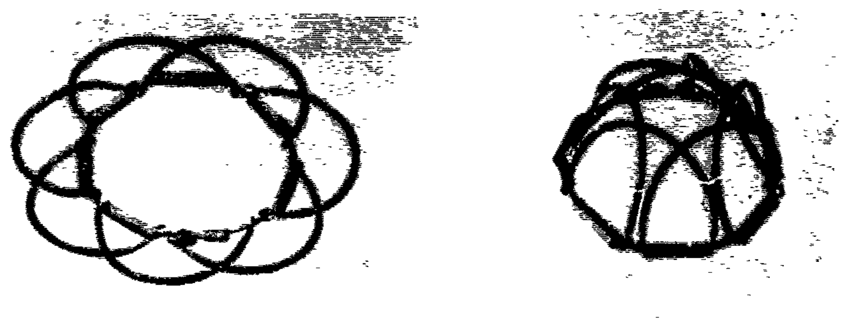


FIGURE 2 SEMI-CIRCULAR ARCH SINGLE STAGE VG STRUCTURE

Degree of Interlacing

The structures shown in Figs. 1 and 2 have a single degree of interlace. (Single degree of interlace refers to the condition of one contact point -- exclusive of the hinge -- between adjacent arches.) The interlaced geometry has two major advantages over the system where nc interlace is present: (1) the resulting framework is a more stable configuration; and (2) the actuation techniques are greatly simplified because of interaction effects.

It is possible to obtain greater interlacing by increasing the number of sides of the base ring polygon, the number of arches, and by varying the hinge location. Qualitatively, however, it appears that the single interlace provides enough stability to enhance the structural characteristics and minimize the actuation requirements.

Manner of Folding

Figs. 1 and 2 show that the arches lie outside of the base ring in a compressed state; this case is referred to as the "outward fold" arch system. It is possible to have the arches fold inside of the base ring, and this case is referred to as the "inward fold."

The inward fold has an advantage of having all of the members, in a compressed state, lie within the base ring. It has the disadvantage in that the arches have definite limitations on their height. The arch height has to be less than the radius of the base ring or complete flattening within the base ring will not occur.

Number of Arches

It was noted above, in the paragraph discussing interlacing, that the number of arches disposed about the ring will influence this parameter. Another aspect of the number of frames is that it will influence the strength of the framework system. Therefore, if the applied load requirements are such that a heavy structure is necessary, then additional strength can be provided by adding more frames.

Base Ring Geometric Variations

The base ring provides a means of geometry variation, but not to a significant extent. It appears possible to design the base ring so that the restraint of the hinges will be minimized.

MULTIPLE-STAGE SYSTEMS

Multiple-stage VG structures are formed by combining single-stage structures into one system. A two-stage structure may be formed by attaching two single-stage structures at their vertex planes. Fig. 3 shows such a two-stage trapezoidal arch system that is compressed and then deployed. It is also possible to form a two-stage system by attaching arches to both sides of the base ring.

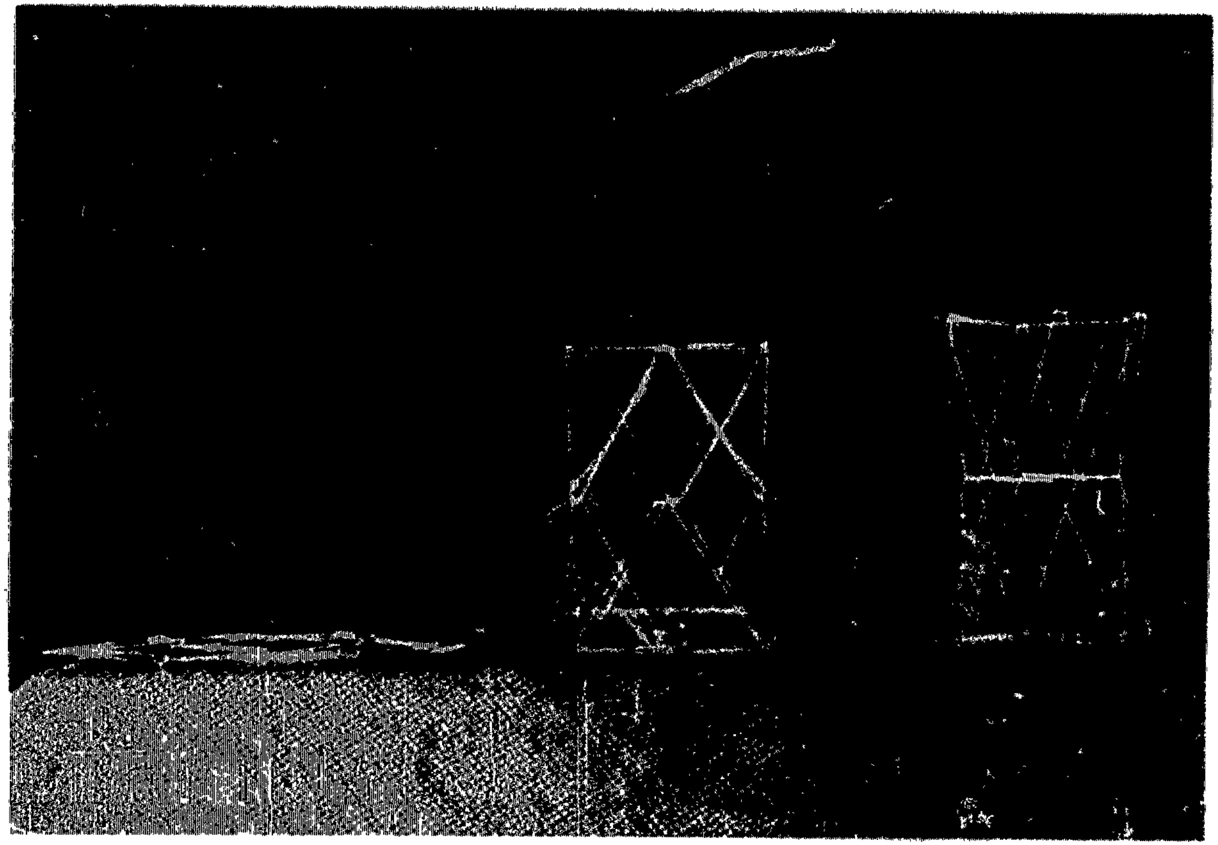


FIGURE 3 DEPLOYMENT OF TWO STAGE VG FRAMEWORK

The multiple-stage feature of the VG structure makes it adaptable to modular type construction. Variations in the structure can be readily accommodated by using different arch configurations or by varying the extent of deployment. This feature will permit variations in volume, surface area, or interior shape as might be desirable.

The extent of variation possible in modular VG structures can be readily seen in Figs. 4 and 5. Fig. 4 shows a triangular arch multiple-stage system at various stages of deployment. The structure, completely compressed, is shown in the left-hand corner. Two states of deployment are shown also in this figure.

Fig. 5 shows a variation of the VG frame structure where a torus is formed. Fig. 5a shows the initial package in a compressed state; Fig. 5b, at partial deployment; and Fig. 5c shows it completely deployed. Detailed examination shows that the arches are essentially triangular, inward-folding with [illegible] interlacing. The curvature to the system is provided by varying the height of the arches that are attached to each base ring such that the developed plane (vertex plane) forms an angle with the plane of the base ring.

SUMMARY

Some of the potential advantages of the VG frame structural system are summarized in Table [illegible].

FRAME AND PANEL STRUCTURES

The arches of the VG frame structure can incorporate panels to provide a more versatile structural system. The marriage of panels and frame arches can take place in several different ways, that is: the panels can be inserted within a given frame; or, the panels may be a separate system but actuated to a [illegible] deployed state by means of the frames.

Figs. 6a and 6b show two configurations which use the panel and frame system. Fig. 6a shows the panel inside the rectangular frames. Interlaced with these rectangular frames is a grouping of triangular frames which provide a means of actuation and deployment. It is not possible to use interlaced panel-filled frames only and provide for actuation by interlacing the frames because of interference between the panels and the frames; the secondary triangular frames are, therefore, provided.

Fig. 6b shows a framework system which is used to deploy a continuous surface area. In this case, the triangular voids between the rectangular panels are filled with auxiliary panels providing surface continuity in the opened state.

While only single stage systems are shown it is possible to connect the frames to each other as was discussed for the multiple stage framework and develop an enlarged structural system. As much design flexibility is possible with this system as is possible with frame networks.

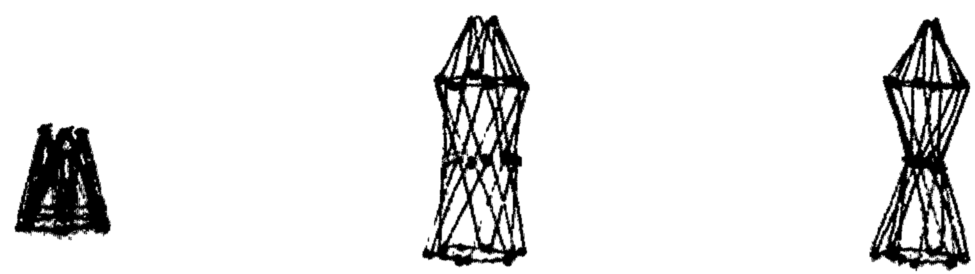
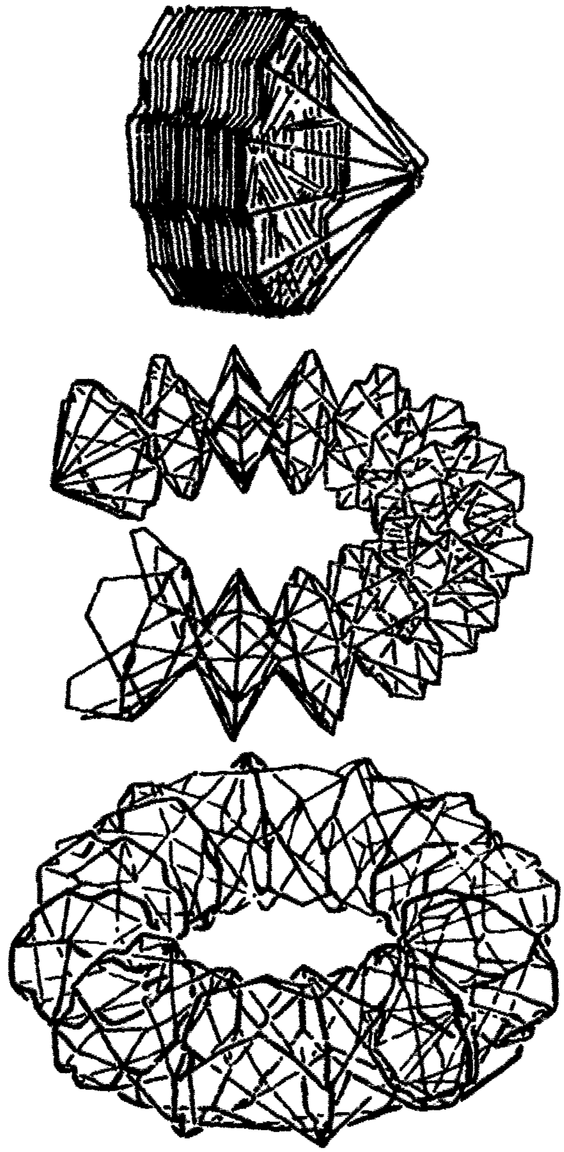


FIGURE 4 TRIANGULAR ARCH – TWO STAGE VG FRAMEWORK



(A)

(B)

(C)

FIGURE 5 TORUS FORMED BY MULTIPLE STAGE VG STRUCTURE

CONCEPT	ADVANTAGES	DISADVANTAGES
General	a Forms a rigid framework to resist applied loads b Large number of configurations attainable by: 1) partial deployment; 2) varying each geometry c Can be integrated with encapsulating containers to provide rigidized structure d Possesses modular features	a Framework not completely enclosed unless special techniques are used b Additional members besides basic arches and base rings required to obtain framework stability
Trapezoidal & Triangular Arches	a Forms triangulated space framework b Volume & area development limited by package size only c Arches can be curved to obtain dome structure d Readily integrated w/panels	a Maximum volume development requires large diameter
Semi-circular Arch	a Forms domed (hemispherical) shape b Arches can be curved to obtain greater dome curvature	a Limited volume expansion b Framework not triangulated; bending will occur c Not readily integrated w/panels
Interlacing Arches	a Increased member stiffness because of lateral support b Few actuators for deployment. (Could be as few as one) c Can be enclosed with panels	a Increased frictional resistance b Increased probability of cold welding
Non-Interlacing Arches	a Lower frictional loads h Less severe enclosure problems	a Less stable b More locking devices c Large number of actuators required

TABLE 1 FRAME ST UCTURES

ENCLOSURE TECHNIQUES

The VG structure provides a framework system which is not capable of complete enclosure or pressure containment by itself. There are many instances where containment is necessary, and investigations were performed to establish the feasibility and limitations of integrating the framework and pressure containment system. The following discussion will emphasize the enclosure of the structure.

LONGITUDINALLY ORIENTED CONVOLUTES

Longitudinally oriented convolutes, (bellows) could be attached directly to the arches, as well as to the supporting ring. The depth of the convolutes at the ring would be zero (fixed diameter ring), and increase gradually to a maximum at the arch vertex plane. Thus, the nvolute skin could be made to expand radially as well as axially and conform to the over-all framework configuration as it opens and closes from disc to cone to cylinder and back again. The convolute skins could be made in two or more layers and provide effective micrometeoroid shielding, incorporated self-sealing techniques, and provide pressurization capability.

SEALS

The panel frame system, Fig. 6, have a means of providing micrometeoroid protection over nearly all of the surface. Techniques whereby the joints can be protected also appear feasible; however, the joints constitute a rather large length which may require sealing in certain instances. A few of the possible methods of achieving this sealing were investigated.

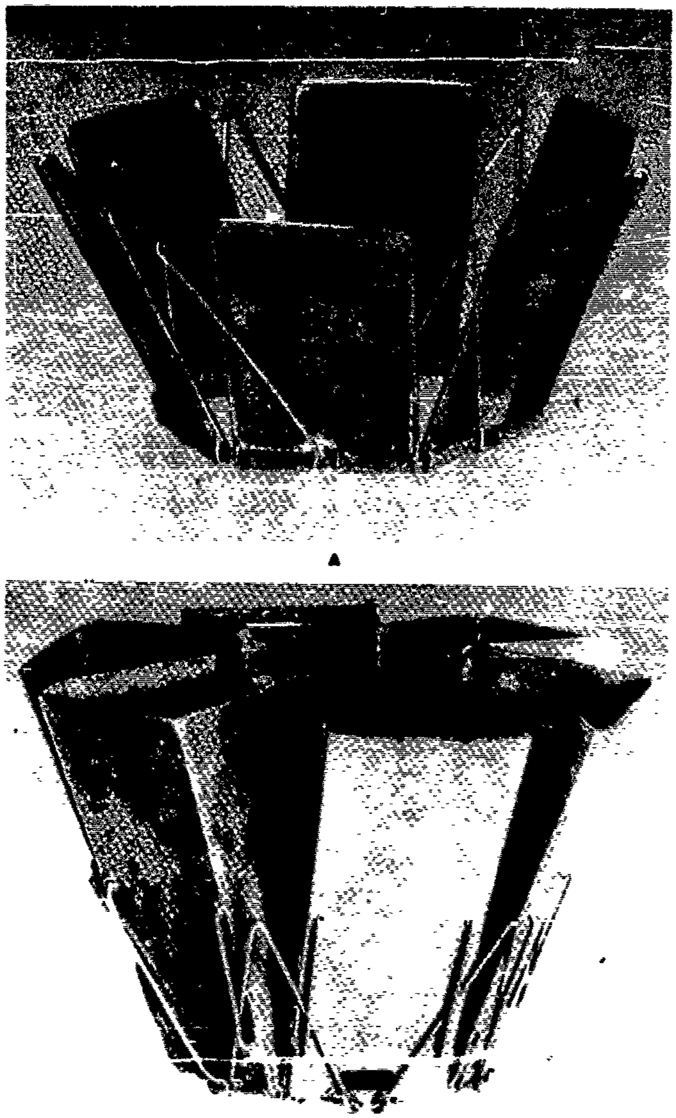
Strip and Inflatable Seals

All hinged seams can be sealed internally with adhesive materials such as RTV 90 silicons or room-temperature vulcanizing butyl rubber, employed to join the sealant to an inner neoprene surface. The effects of absorption of ultraviolet radiation may require the application of thin protective metallic coating to the sealant, such as aluminum, to minimize degradation of the polymetric sealants.

In a panel configuration the self-sealing materials can be shielded from the relatively low-level radiation anticipated for an orbit below 500 miles (with the exception of unpredictable solar flares) by the outer sandwich structure. The variable of importance in this design is the temperature of the self-sealing composite materials. This can be accommodated by appropriate external thermal control coatings.

Continuous Bladder Seals

A more positive method of sealing the entire system is to inflate a pressure-tight bag, inside or outside of the structure, after it has been deployed and, thus, eliminate the possibility of leakage at seal intersections or overlaps. One possible design would be a bag fabricated with expansion convolutions and bonded to the inner walls of the VG panel-frame structure.



A

B

FIGURE 6 PANEL-FRAME SYSTEMS

This method will require rub-strips over rough surfaces and the use of corner-filling materials in deep recesses for bag protection, but a positive seal is assured even under such adverse conditions as docking operations, coupling, and thermal expansion and contraction.

CONCLUSIONS

The investigations have primarily emphasized the conceptual aspects of the VG structure. Within this limited investigation it was found that a large number of different structural geometries are possible, and it appears that the concept can be implemented to form full scale structural systems.

The limited scope of the study has not permitted the detailed examination, design and fabrication of more exact models or full-scale systems. The next logical step is the implementation of the concept by an engineering study ultimately providing hardware. This will point up the uncertainties and problems associated with this system.

The integration of the rigid framework VG structure with the inflatable structure offers an approach whereby rigidity can be developed in the system and internal pressurization can be maintained for life support.

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TELESCOPING PRESSURIZED STRUCTURES AND SEALS

By W. Bandaruk, D. G. Younger, and N. E. Quackenbush

INTRODUCTION

The primary advantage gained by the use of the expandable structure concept for manned spacecraft application is simply the fact that large structures or elements of structure can be packaged into compact volume for boost from earth. A number of schemes have been proposed for increasing the volume or shape of the structure, once it is carried into orbit. These may include methods for unfolding panels, telescoping, or pure inflation of structural components[1].

The achievement of high reliability in the deployment of an expandable spacecraft structure is, of course, desirable, and this can best be obtained by utilizing simple methods of expansion and sealing. In addition, a method of deployment free from complexity should also provide easier control of re-collapse of the structure. Ease of re-collapse would be a desirable feature from the standpoint of gathering all available material around the crew members during solar flares or meteoroid storms. Figure 1 shows an expandable spacecraft configuration that can operate as an induced gravity vehicle (rotating space station). The configuration displays the telescoping concept. For this vehicle, the long expandable booms provide both the structural continuity and/or the lengthy passageways to the outer modules. The configuration resembles that of a rotating dumbbell when fully deployed. As shown, the spacecraft consists of a rigid hub and laboratory modules, which are also expandable, located at the opposite extremities of the configuration[1].

The deployment of the space station thus described can best be performed through pneumatic erection. That is, the environmental air required within the station can be introduced from gas containers by means of a regulated system to provide controlled expansion. Although the study to be described herein is primarily that of seal concepts, a brief discussion of a self-regulating pneumatic system is included in the Appendix.

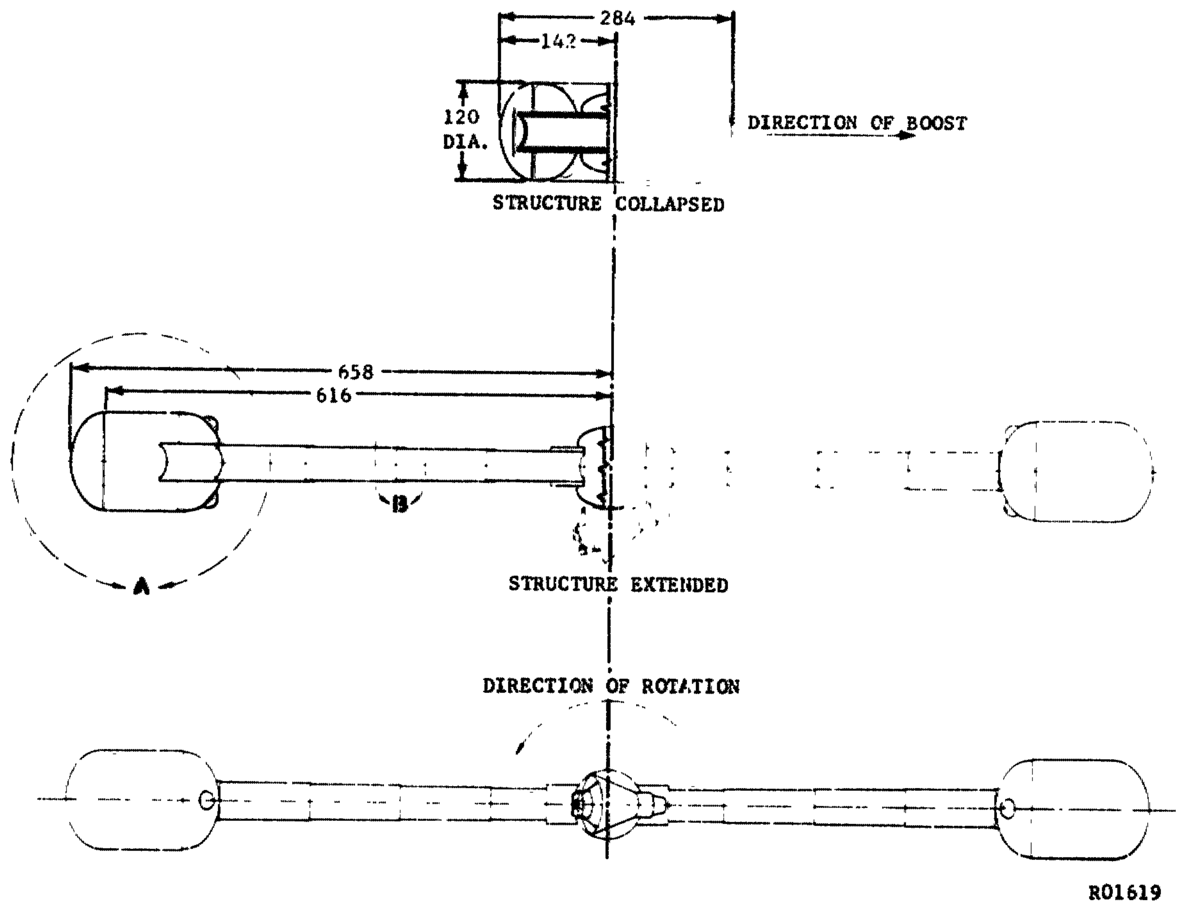

FIGURE 1. EXPANDABLE STRUCTURE CONCEPT FOR ARTIFICIAL GRAVITY SPACE STATION

ADVANCED TELESCOPING DESIGN CONCEPTS

In a study of telescoping structures having hermetic integrity, four telescoping concepts were advanced which incorporated relatively elementary sliding movements adaptable to the mating sections of rigid construction. In each design, Teflon rollers have been included at the section terminals to insure both a smooth, low-resistance telescoping motion and a medium for transmitting bending moments that may develop during extension. At full deployment, each of these designs embraces a means for transmitting mechanical loads.

Design Concept Number 1, shown in Figure 2, reveals a scheme in which bending moments are transferred through the rollers at the extremities of each cylindrical section. Axial compressive loads are resisted by several adjustable locking brackets positioned manually, as shown, around the inner periphery of the joint. Tensile loads acting through the joint are transmitted through the deformed O-rings, the guided spacer sleeve, and the terminal roller brackets of each wall section. During deployment, leakage is held to a minimum by two O-rings rolling in the angular space between the telescoping sections. At full deployment, a static seal is developed by the deformed O-rings compressed into place.

Design Concept Number 2, presented in Figure 3, transmits bending moments and tensile loads through the medium of mated tapered surfaces. Compressive loads are resisted by manually positioned bolts that hold the tapered surfaces in place. During extension of the cylindrical sections, a positive seal is accomplished by a flexible membrane attached between the sections as shown. The flexible membrane feeds along between the cylinders during the extension. At full deployment, the membrane is assisted in making the static seals by two O-rings integrally positioned between the mated tapered surfaces.

Design Concept Number 3 utilizes a Belleville spring to rigidize the joint and transmit bending loads after full extension. This concept is shown in Figure 4. Tensile loads are transferred through the combination of the collapsed bellows, the springs, the spacer rings, and the rigid stops at the section terminals. A medium for handling compressive loads is not indicated in the figure; however, an automatic toggle locking feature could be employed. The positive seal during and after deployment is provided by the welded diaphragm bellows mounted between the sections.

Design Concept Number 4 is shown in Figure 5. The concept employs a set of cylindrical metal sleeves and coated fabric diaphragms bonded between the sleeves. The bending moments in this joint are transmitted by rollers mounted at the section extremities. Tensile loads are transferred through the thicker metal sleeve that is mounted so as to intercept the approaching ring-roller assembly of the adjacent cylindrical section. The coated fabric performs as a rolling diaphragm between the metal sleeves and seals the telescoping sections during deployment. After full extension, a more positive seal is attained by the flexible seal placed along the line of contact of the thicker metal sleeves and the ring-roller assembly.

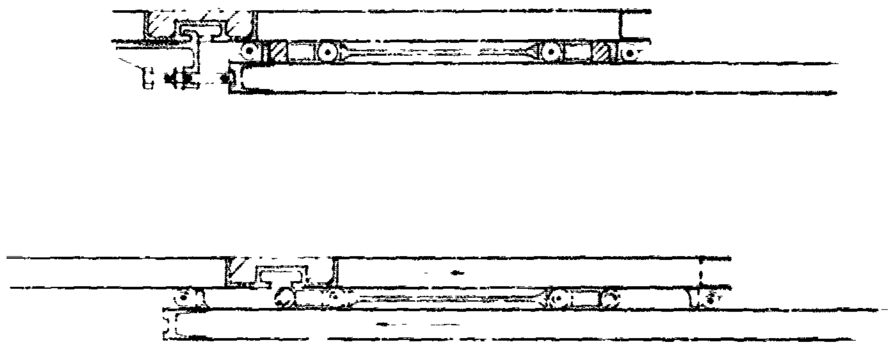

FIGURE 2. DESIGN CONCEPT NO. 1, USING O-RINGS AND SPACER SLEEVE

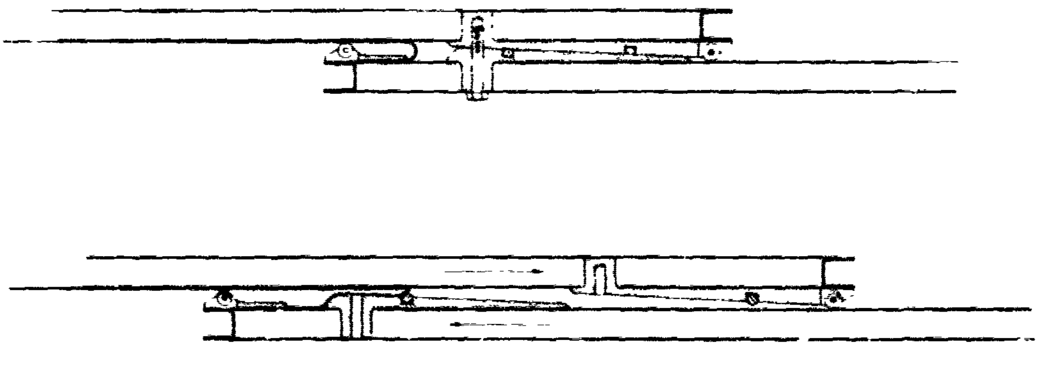

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FIGURE 3. DESIGN CONCEPT NO. 2, EMPLOYING TAPERED SURFACES, FLEXIBLE MEMBRANE, AND O-RINGS

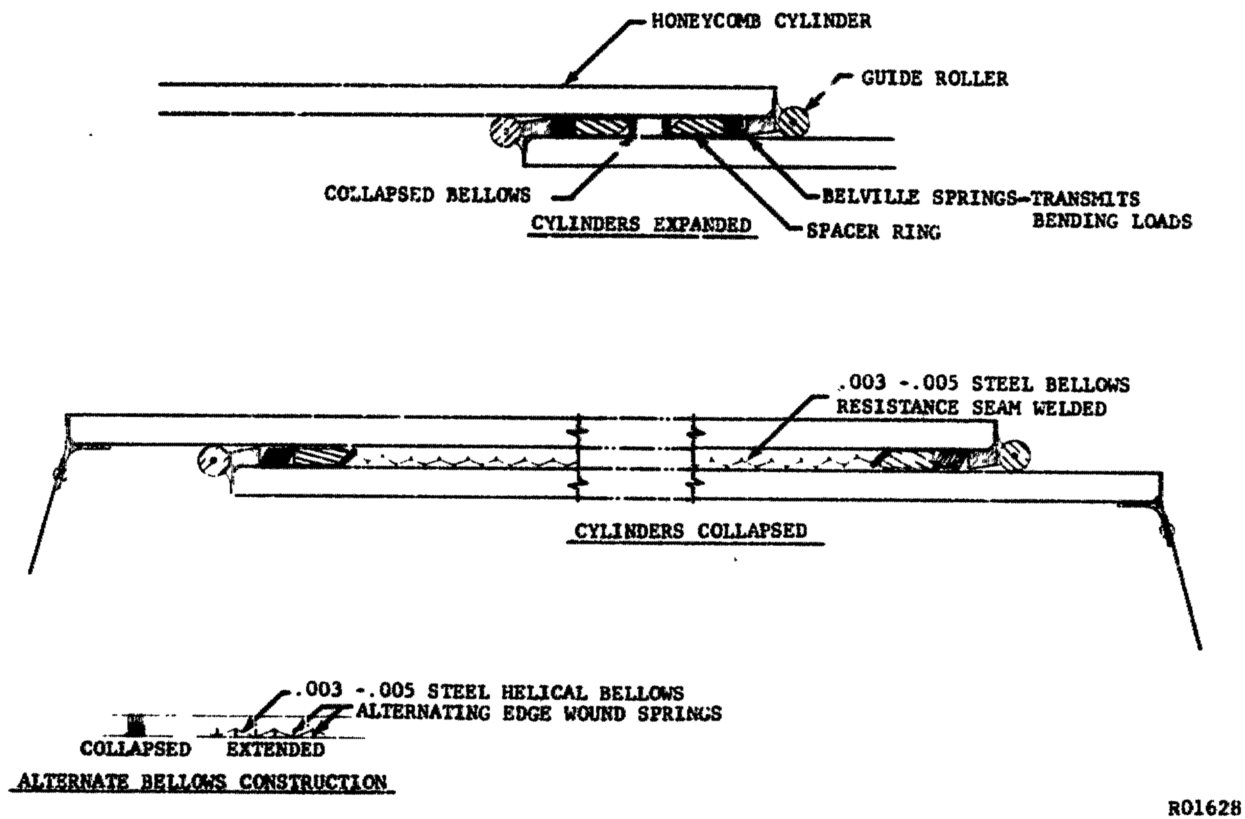

FIGURE 4. DESIGN CONCEPT NO. 3, UTILIZING BELLEVILLE SPRING AND BELLOWS

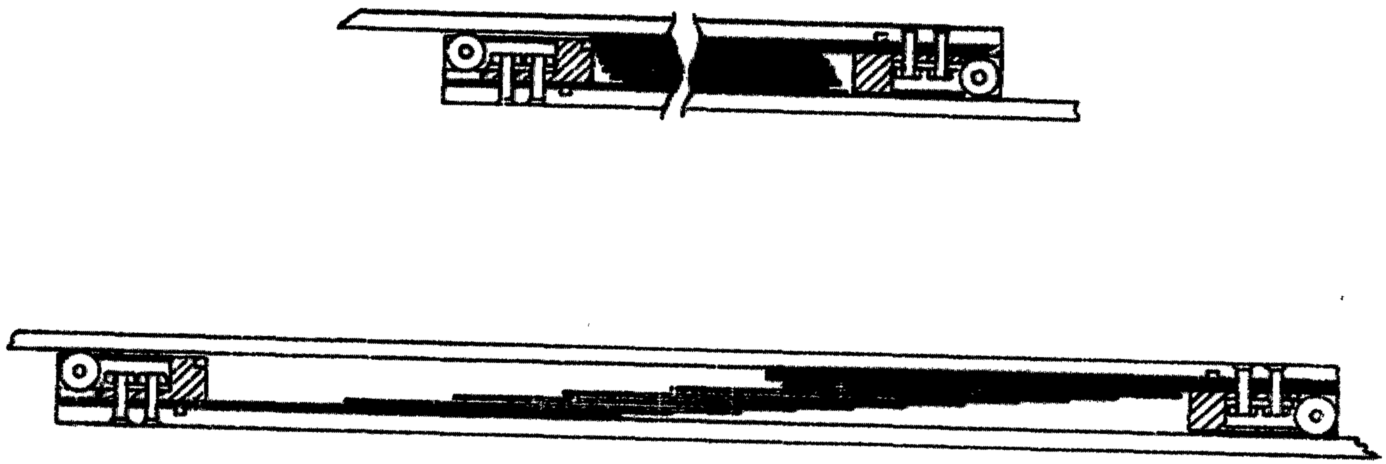

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FIGURE 5. DESIGN CONCEPT NO. 4, INVOLVES COATED FABRIC DIAPHRAGM BONDED BETWEEN TELESCOPING CONCENTRIC ALUMINUM SLEEVES

Design Concept Number 4 was selected for development and demonstration. This concept appears to have the inherent advantage that it requires little or no force to extend or collapse the configuration. Thus, the scheme should display negligible resistance to the movement of the mating sections. Furthermore, the metal-sleeve rolling diaphragm can be easily contained and protected within the small annular space between the mating telescoping sections.

MATERIALS INVESTIGATION FOR FLEXIBLE DIAPHRAGMS AND SEALS

The prime materials consideration for the flexible diaphragm was a fabric impregnated with an elastomer. A number of synthetic textile fibers were studied; however, on the basis of strength and commercial availability, consideration was limited to utilization of Dacron, glass fibers, Nylon, and Fortisan. Further screening of these textile materials as to how they are affected by atmospheric conditions limited the choice of fabrics. Moisture has an appreciable effect on Nylon and Fortisan, since they lose 10 percent and 15 percent, respectively, of their dry strength. On the other hand, Dacron and glass fibers are not significantly affected, for this application, by moisture conditions.

The elastomers considered as candidates for impregnating these fabrics were silicones and fluorinated polymers[2]. These elastomers were selected primarily because of their low temperature properties and low surface friction characteristics.

As a result of these material considerations, four flexible membranes were investigated as seals between the concentric aluminum sections of the simulated space vehicle. These flexible membranes were obtained from Minnesota Mining and Manufacturing Company and are as follows:

(1) SRGA-0208, a 3.7-ounce glass fabric impregnated with a silicone elastomer and aluminum vapor coated on one side.

(2) An experimental 3.1-ounce (Type 116) glass cloth impregnated with Teflon and aluminum vapor coated on one side.

(3) FG-11, a 3.7-ounce glass cloth impregnated with a fluorinated elastomer (Fluorel).

(4) FD-8, a 3.2-ounce Dacron fabric impregnated with Fluorel.

The first flexible membrane investigated (Y-950) was not satisfactory, because the glass fabric could not withstand repeated 180-degree bending and sliding over itself. Leaks developed through the silicone elastomer, and the vapor deposited aluminum film developed voids from the abrasive action of the film against itself. In addition, no successful void-free adhesive system was found for bonding silicone to aluminum (fabric to metal ring). The usual RTV silicones did not cure when in contact with the coated

fabric, and further testing on other types of adhesive was not considered worthwhile, because other physical properties of the preimpregnated fabric were not acceptable.

The second and third flexible membranes, consisting of Teflon and Fluorel-impregnated glass cloth, had the same major difficulties found in the silicone-glass fabric. Principally, the glass fabric failed during the repeated 180-degree bending and movement over itself.

Predominant effort in the materials investigation was centered on the fourth item--namely, the Dacron fabric impregnated with Fluorel (FD-8). Table 1 shows the physical properties of this flexible material.

FD-8 had good continuity under repeated flexing and abrasion; however, it was difficult to make and maintain a gas-tight seal between the aluminum sections. The cyanoacrytate, Eastman 910, with the Budd activator was found to be an excellent adhesive system, but it did not possess the desired flexibility and had a tendency to pull the Fluorel coating from the fabric under the peel loads encountered. Bonding restraining tabs of a lightweight Dacron fabric, to carry the load between sections, reduced this problem to a minimum.

The developed procedure for bonding the Dacron fabric impregnated with Fluorel rubber was as follows:

(1) Strips of FD-8 were cut, and a small wedge was trimmed from each end to obtain a slight tapered effect. This was required to compensate for the decrease in diameter of each successive aluminum ring.

(2) The bands of FD-8 were bonded to the aluminum rings, using Eastman 910 and Budd GA-1A. The ends of the FD-8 strips were bonded together with Fairprene cement 5159 (a two-part fluorinated elastomer cement) in making an overlap joint of FD-8.

(3) The Dacron fabric restraining tabs were bonded with EC-1357 (neoprene cement).

(4) The bond between the FD-8 and aluminum rings was sealed with ED-2174 (nitrile phenolic cement).

(5) The bond or joint was checked by pressurizing the telescoping structure and brushing soap solution around the openings between the rings.

TABLE 1

PHYSICAL PROPERTIES OF DACRON
IMPREGNATED WITH A FLUORINATED ELASTOMER (FLUOREL)

Property	Federal Specification CCC-T-191B Method Number	FD-8*
Base fabric		Dacron
Base fabric weight (oz/sq yd)	5041	3.2
Base fabric thickness		0.0065-inch
Coated fabric weight (oz/sq yd)	5041	8.0 ± 0.5
Coated fabric thickness		0.009 ± 0.001-inch
Tensile strength (lb)	5100	Warp: 200 Fill: 180
Tear strength (lb)	5134	Warp: 12 Fill: 10
Bursting strength (lb/sq in.)	5122	340
Crease resistance		10-pound roller, no cracking
Stiffness (bending moment, in-lb)	5202	Warp: 0.00620 Fill: 0.00360
Abrasion resistance CS-17 wheels, 500 gm weights Taber wear index: $\frac{\text{wt loss (mg)} \times 1000}{\text{100 cycles}}$ =	5306	150
Coating adhesion (lb/in. width)	5970	15
High-temperature stability		100 hours at 400°F, no change in physical properties
Low-temperature flexibility (2 hr exposure, no cracking)	MIL-C-20696	-100°F
Weather resistance		1-year outdoor exposure in industrial atmosphere, no change in physical properties
Fungus resistance	MIL-I-7444B	No growth
Permeability (fl oz/sq ft/24 hr)	ASTM-D-814-55	0.15
Flame resistance	5902 flaming time (sec)	Slightly flammable

* All values shown are average and obtained from Minnesota Mining and Manufacturing Company Bulletin F3.

THE DEVELOPED TELESCOPING CONCEPT

An exploded and assembled view of the demonstration configuration is shown in Figure 6. As can be seen, the developed telescoping model consists of the following items:

(1) Inner and outer cylindrical section, with dome closures welded at the closed ends. The test-bed cylinders and domes are fabricated from 6061-T6 aluminum alloy selected primarily for ease of forming and weldability.

(2) An aluminum ring and Teflon roller assembly is shown attached to the mating open ends of each cylinder. The ring and roller assembly serves to impart alignment and smooth movement to the mating cylinders. The ring of the inner cylinder is mechanically attached to the outer surface of the cylinder, while the ring of the outer cylinder is similarily attached to the inner surface of the cylinder. The rollers on each ring bear respectively against the other cylinder.

(3) A set of concentric aluminum-alloy 6061-T6 sleeves are located in the annular space between the mating cylinders. The seal is accomplished by a series of coated fabric diaphragms located among the concentric sleeves. The outer and inner sleeves of the seal concept are attached, respectively, against the outer and inner cylinders and are attached in place by the ring attachment.

The positioning of the concentric metal sleeves and the flexible rolling diaphragm are shown in the small cutaway view at the lower right-hand corner of Figure 6. Here the telescopic cylinders are fully deployed. The inner and outer sleeves, which are longer than the intermediate sleeves, are shown in black to emphasize their position between the rings and cylinders. All sleeves are 0.04-inch thick, except the outer sleeve which is 0.125-inch thick. The thicker sleeve is located so that it intercepts the approaching ring of the inner cylinder and also functions as a mechanical stop restricting further movement. When this seal concept is subjected to a differential pressure of 15 psi, the concentric sleeves receive a net pressure loading that is external, and structural instability (buckling) considerations dictate at least a 0.04-inch sheet thickness for the sleeves. After the telescopic cylinders are completely deployed and fully pressurized, the metal sleeves and rolling fabric diaphragm are packed concentrically into the small remaining annular space, as shown in the cutaway view at the lower right.

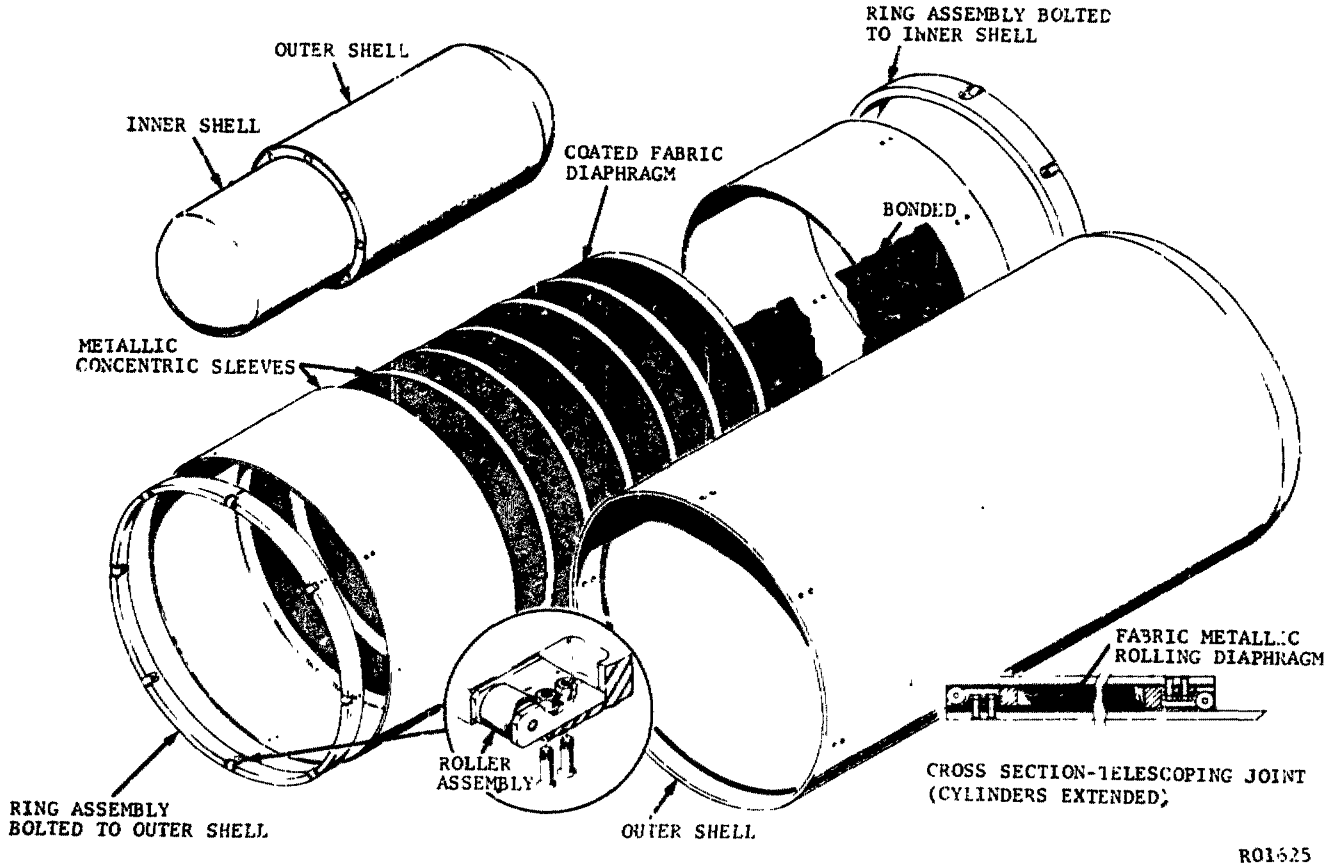

FIGURE 6. DEMONSTRATION SPECIMEN SHOWING SEAL CONCEPT FOR TELECOPING CYLINDERS

The developed components of the demonstration model are shown in Figure 7. The seal concept, reduced to practice, is shown in the center being held in the extended position. The dimensions in detail and the materials utilized in developing this seal concept are given in Table 2. Basically, the fabricated seal consists of seven metal sleeves and six coated-fabric rolling diaphragms weighing a total of 15.56 pounds. All metal-to-metal fits between sleeves, rings, and cylinders were held to close tolerance, and light tapping with a rubber mallet was sufficient to assemble these parts. To further insure sealing, washer-type seals were utilized underneath each of the Phillips-head screws attaching the seal and roller assembly to the inner and outer cylinders.

TESTING SETUP

The developed configuration, assembled and instrumented prior to pressurization, is shown in the vertical position in Figure 8, contracted position, and in Figure 9, extended position. The upright position was utilized in all the tests. The testing was accomplished as follows:

(1) Air pressure was introduced through an inlet provided at the lower extremity of the specimen, and this inlet pressure was monitored with a Gunkle pressure gage, as shown in the figure.

(2) Pressures developed within the configuration were precisely recorded by means of a Statham pressure transducer sensing static pressure through a fitting inserted at the uppermost extremity of the moving cylinder.

(3) Outputs from the transducer were recorded by a Sanborn Model 297 direct-write oscillograph used in conjunction with an Owens Laboratories amplifier (and bridge control panel) with power supply.

(4) Vertical displacement of the inner cylinder was sensed by a cord attached (as shown) to the moving cylinder, which rotated a Borg linear potentiometer.

(5) The output of the potentiometer was then recorded on a second channel of the oscillograph.

TESTS AND RESULTS

Using the test setup described above, the following series of tests were conducted:

(1) Determination of the frictional characteristics of the model

(2) Evaluation of integrity to pressure forces

(3) Evaluation of leakage rate associated with the developed seal

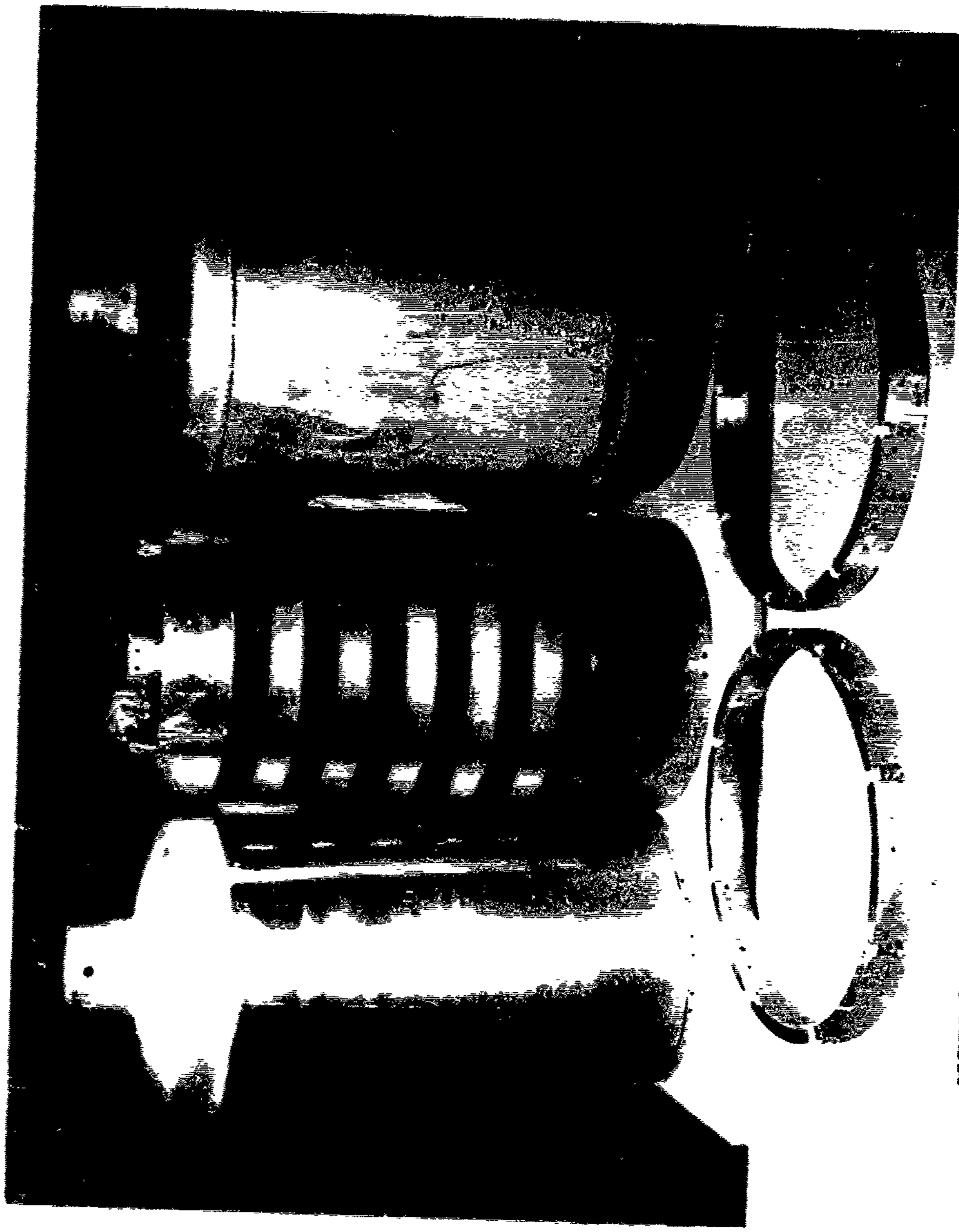

FIGURE 7. DISSASSEMBLED TEST FIXTURE AND COATED-FABRIC ROLLING DIAPHRAGM SEAL

TABLE 2

MATERIAL DETAILS AND DIMENSIONS OF SEAL CONCEPT

Item	Dimensions (Inches) and/or Material		
1. METAL SLEEVES	Inside Dia.	Length	Thickness
a) Outer	19.250	8.250	0.125
b) Inner	18.000	8.000	0.040
c) Five intermediate	18.215	6.000	0.040
	18.420	6.000	0.040
	18.625	6.000	0.040
	18.830	6.000	0.040
	19.035	6.000	0.040
	6061-T6 Aluminum alloy, fusion welded		
2. COATED FABRIC	3.2-oz Dacron fabric coated with a cured black Fluorel* elastomer (3M, FD-8)		
a) Weight	8.0-oz/sq yd (coated fabric)		
b) Thickness	0.009 in. (coated fabric)		
c) Tensile strength	200 lb/in. (warp) - 180 lb/in. (fill)		
3. ADHESIVE	Eastman 910 (cyanoacrylate) to bond coated fabric to aluminum		
4. ACCELERATOR	Budd GA-1A surface activator to promote the polymerization of Eastman 910		
5. CEMENTS			
a) Fabric overlap bond	DuPont Fairprene cement 5159; a two part, Viton (fluro-elastomer) base, solvent cement used to make overlap bonds between the segments of coated fabric		
b) Fillet seal	3M, EC-2174 nitrile-phenolic. solvent cement used to fillet seal the bonded edges between the coated fabric FD-8 and aluminum		
c) Cloth bonding	3M, EC-1357 neoprene contact cement used to bond Dacron supporting tabs		
6. CLOTH	Dacron, 0.005-inch thick, fine-weave		

*Fluorel is a fluorinated polymer with high temperature and high chemical resistant properties.

FIGURE 8. TEST SETUP FOR DEMONSTRATION OF SEAL FOR TELESCOPING CYLINDERS - CONTRACTED POSITION

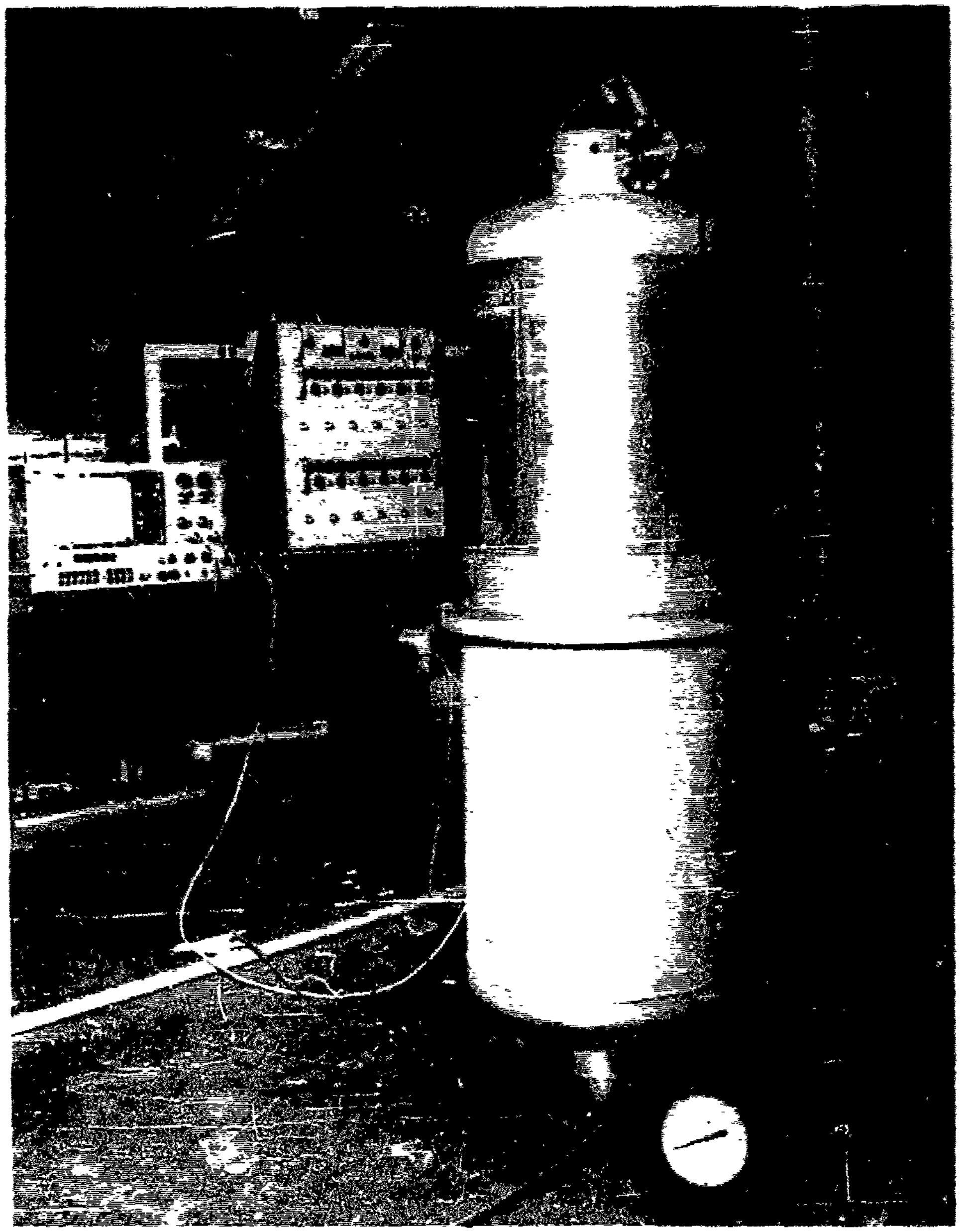

FIGURE 9. TEST SETUP FOR DEMONSTRATION OF SEAL FOR TELESCOPING CYLINDERS - EXTENDED POSITION

Determination of Frictional Characteristics

The first series of tests was performed to determine the frictional characteristics of the developed model. The friction tests involved extending and contracting the cylinders approximately 20 cycles. Measurements were made of the internal pressure and increment of expansion during the extension half and contracting half of the cycle. No significant differences existed between data recorded for each of the pressure cycles. The relationship found to exist between pressure and cylinder extension during each of the telescoping cycles is presented graphically in Figure 10.

Obtaining the data began by first slowly pressurizing the collapsed configuration. As the curves show, the cylinder extension did not initiate until the internal pressure reached a level of 0.53 psig. This is the pressure required to overcome (1) the initial friction in the system, (2) the weight of the moving cylinder (105.70 pounds), (3) the weight of the transducer and fittings (15.46 pounds), and (4) the weight of the seal assembly (15.56 pounds). A slow rise in pressure was required to attain continuing deployment until an extension of about 3.5 inches was reached at a pressure of 0.55 psig. Then, throughout the remaining portion of the deployment, this pressure level was maintained at approximately a steady plateau, except for the last inch of extension when a slight rise in pressure was required. Three distinct drops along the extension cycle indicated momentary drops in friction. These periodic drops in friction were not evident, however, in recordings of deployment versus time, as shown in Figure 11. This figure also shows that two different ates of extension and contraction were employed; however, neither of the time rates affected the data displayed in these graphs.

At the completion of deployment, the pressure was increased momentarily to 5.0 psig and then permitted to drop slowly by slightly opening the inlet air line to ambient pressure. The inner cylinder remained fully extended until the pressure decreased to approximately 0.50 psig, and at this pressure level the inner cylinder began to collapse slowly at essentially constant pressure. The constant pressure plateau during contraction was due to the combined weight of the descending inner cylinder and attached components. During the descent, six distinct dips in the pressure recording were observed, as shown in Figure 10. These dips are considered as indication of regions of increased friction drag in the seal. It is evident that these periodic increases in friction are the result of the seal movement, as each metal sleeve and the attached rolling fabric imparted its relative movement to the adjacent sleeve. Again, these regions of increased friction did not display any effect on the tests of contraction versus time, as shown in Figure 11.

The frictional force inherent in the seal concept can be determined from the data presented in Figure 10. The determination can be made from the difference between the force required to extend the cylinder (0.55 psig) and the lift-off pressure required (0.505 psig). The pressure difference is found to be 0.045 psig. Since the effective piston area of the inner cylinder is about 276 square inches, the above pressure difference caused by friction results in a total force required of 12.4 pounds. In terms of force per peripheral inch of seal, the friction force is equal to $\frac{12.4}{18.75\pi}$. Calculated,

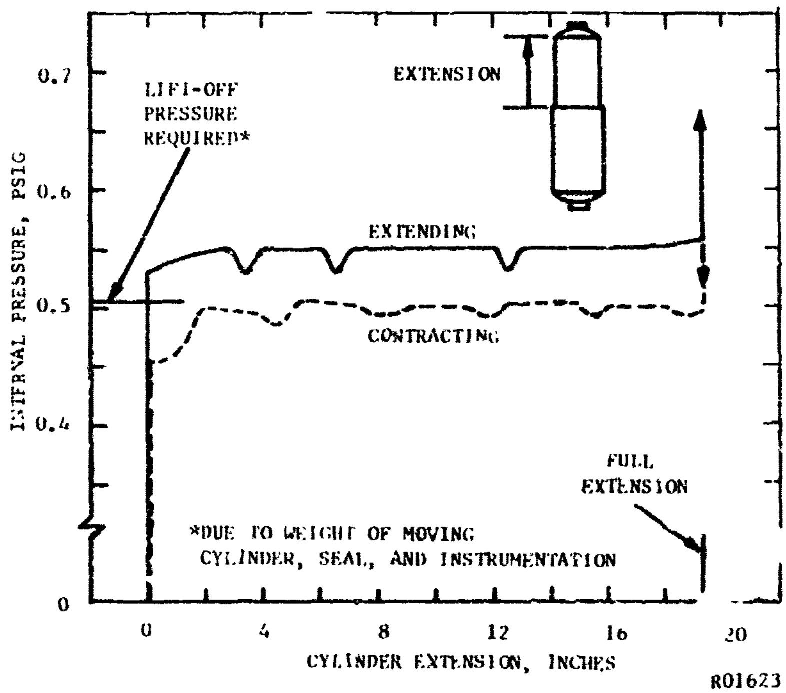

FIGURE 10. INTERNAL PRESSURE REQUIREMENTS FOR THE ROLLING DIAPHRAGM SEAL CONCEPT

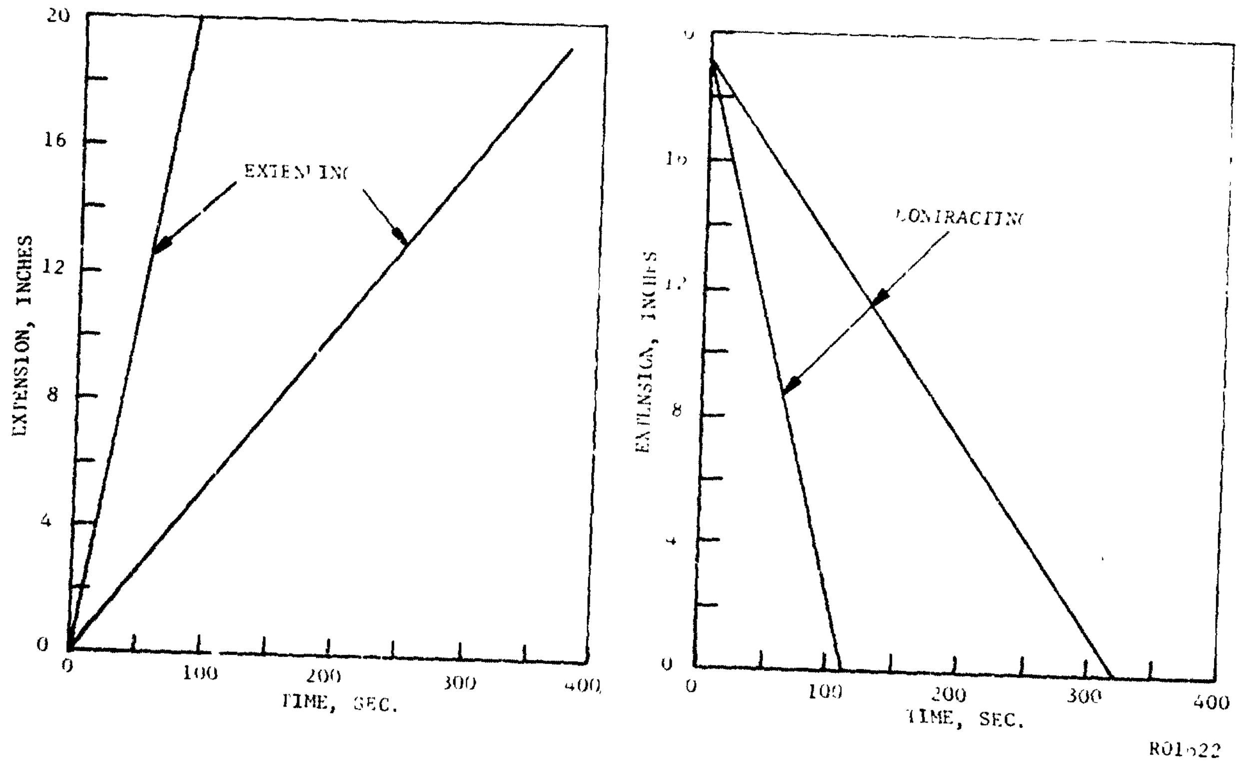

FIGURE 11. EXTENSION AND CONTRACTION RATES IN TELESCOPING CYLINDER DEMONSTRATION

this equals 0.21 pounds per inch. The diameter (18.75 inches) utilized in this calculation is the mean diameter of the seal configuration. Thus, the inherent friction in the seal model developed in the program is very low, and in the weightless environment the only force required to extend the telescoping sections employing this seal would be that necessary to overcome the friction force per lineal inch of 0.21 inch. In a similar manner, the frictional force that must be exceeded during contraction is computed to be 0.12 pound per peripheral inch of seal except during the last two inches before complete closure. At this point, the friction force rises to 0.23 pound per inch.

Evaluation of Integrity to Pressure Forces

For these tests, the static seal which ordinarily provides the seal after full extension (see Figure 6) was purposely eliminated. Thus, the coated fabric, rolling-diaphragm seal was subjected to the full pressure loading. The telescoping model was fully extended under internal pressure, and the pressure level was raised to 8.0 psig. Gross leakage checks were made, but no adverse leakage was detected. Next, the internal pressure was increased to 15.0 psig, and again no adverse leakage was noted. The increased pressurization from 8 to 15 psig apparently had no appreciable effect on the integrity of the fabric-metal bonded joints.

Evaluation of Leakage Rate Associated with the Developed Seal

This series of tests was performed to evaluate the leakage rate associated with the overall seal concept. Again during these tests the static seal which provides the seal after full extension was not used, so that the coated-fabric rolling diaphragm seal was exposed to the full pressure loading. The telescoping configuration was fully deployed under internal pressure, and the pressure level was increased to 8.0 psig. At this point the leak-rate (or permeability) test was conducted. This test consisted of closing the inlet air source and recording the internal pressure level retained within the model as a function of time. The recorded data are shown in Figure 12. As can be seen by the curve faired through the data, approximately 200 minutes elapsed before the pressure dropped to one-half the initial level. The permeability test was further continued by increasing the internal pressure to 15.0 psig, and the leak rate test was repeated. The data at this level of pressurization are also shown in Figure 12. As can be seen, the effective permeability of the seal remained essentially unchanged.

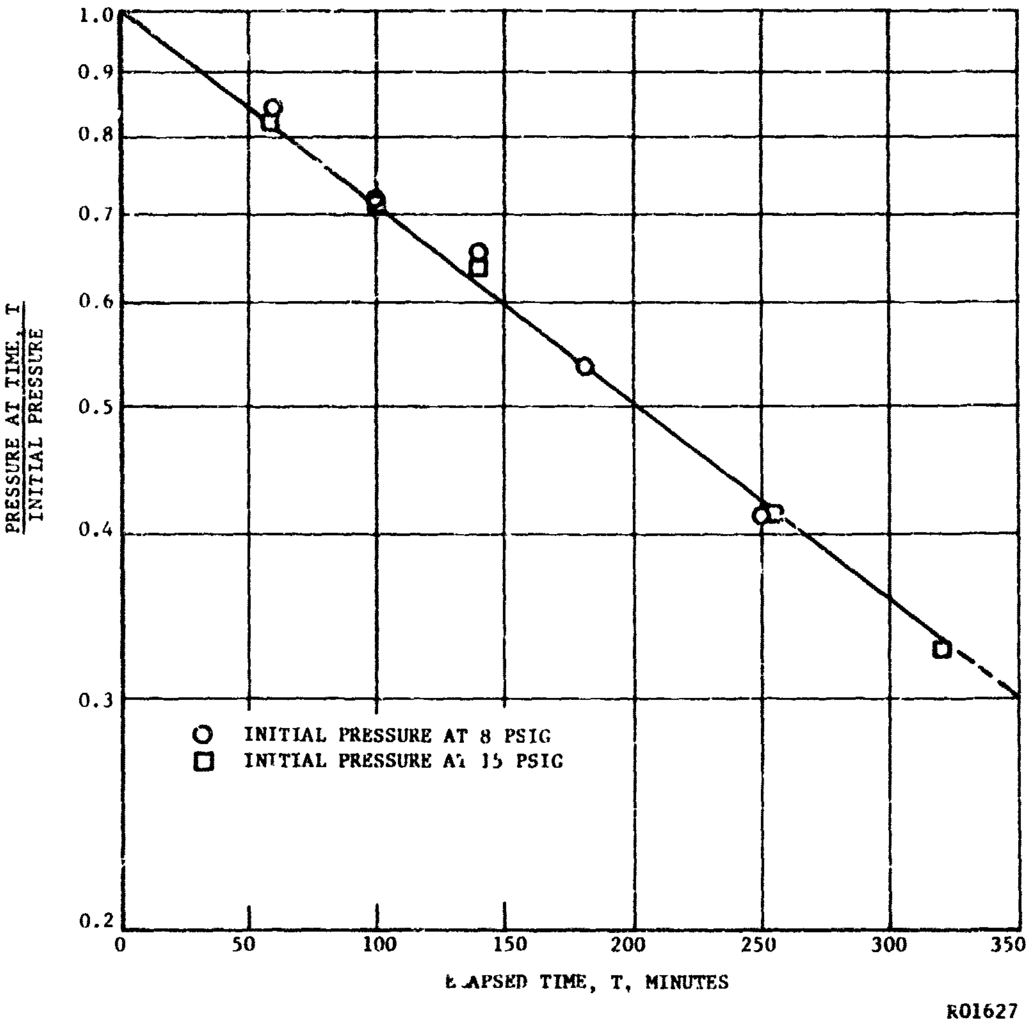

FIGURE 12. PRESSURE-TIME RELATIONSHIP FOR ROLLING DIAPHRAGM SEAL CONCEPT

APPENDIX

VARIABLE PRESSURE PNEUMATIC SYSTEM

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The following is a description of a variable pressure system for pneumatic erection of telescoping structures.

PRINCIPLE OF OPERATION

The pneumatic system is shown schematically in Figure 13. The gas container is sealed by means of two redundant squib valves. When an electric signal actuates the valves, gas discharges through a reduction orifice into the volume inside the boom. The reduced gas pressure activates the release mechanism and extends the structure. The system is self regulating in that it provides sufficient pressure during extension and provides considerable increased pressure in case of unexpected jamming.

GAS CONTAINER

The relationship between the length of the boom and the required volume of the gas container is given in Figure 14. The diagram was calculated assuming adiabatic expansion of the gas into a telescoping boom of 5 feet diameter. The parameter P_1/P_2 is the ratio between the initial pressure in the gas container and the nominal pressure needed to extend the structure. From the diagram, we find that for erecting a telescoping boom 100 feet long (5 feet diameter), using a conservative pressure ratio of 1,000, the required volume of the gas container is 14.05 cubic feet. The corresponding dimension of a equilateral cube is 2.4 feet. The required volume is even smaller for higher pressure ratios.

The final choice of the pressure ratio will depend upon: (1) pressure P_2, which is the sum of the experimentally determined nominal pressure required for extending the telescoping boom and a safety margin; and (2) initial pressure that determines weight of the gas container.

SQUIB VALVES AND REDUCTION VALVE

Actuation of a highly reliable squib valve starts an automatic sequence of events of structure deployment.

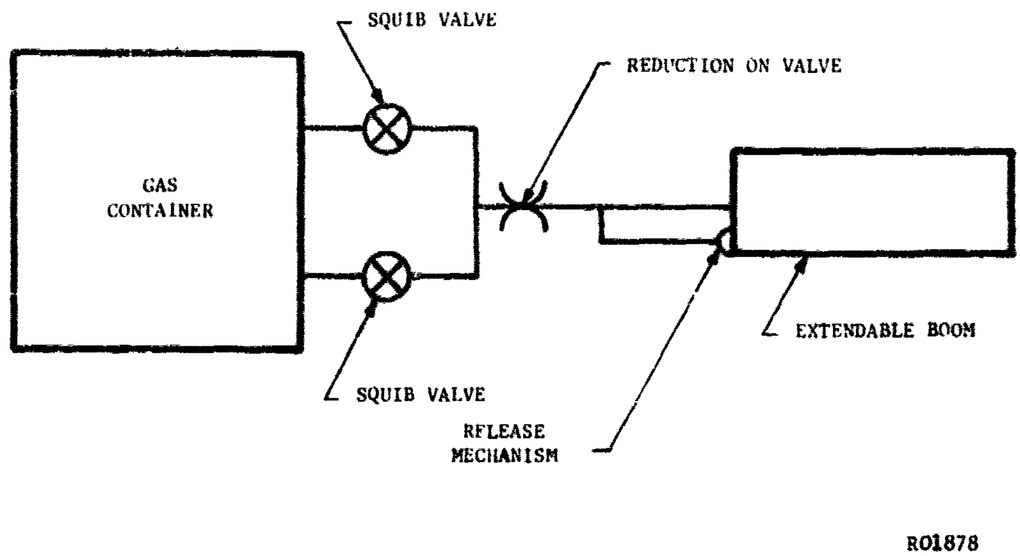
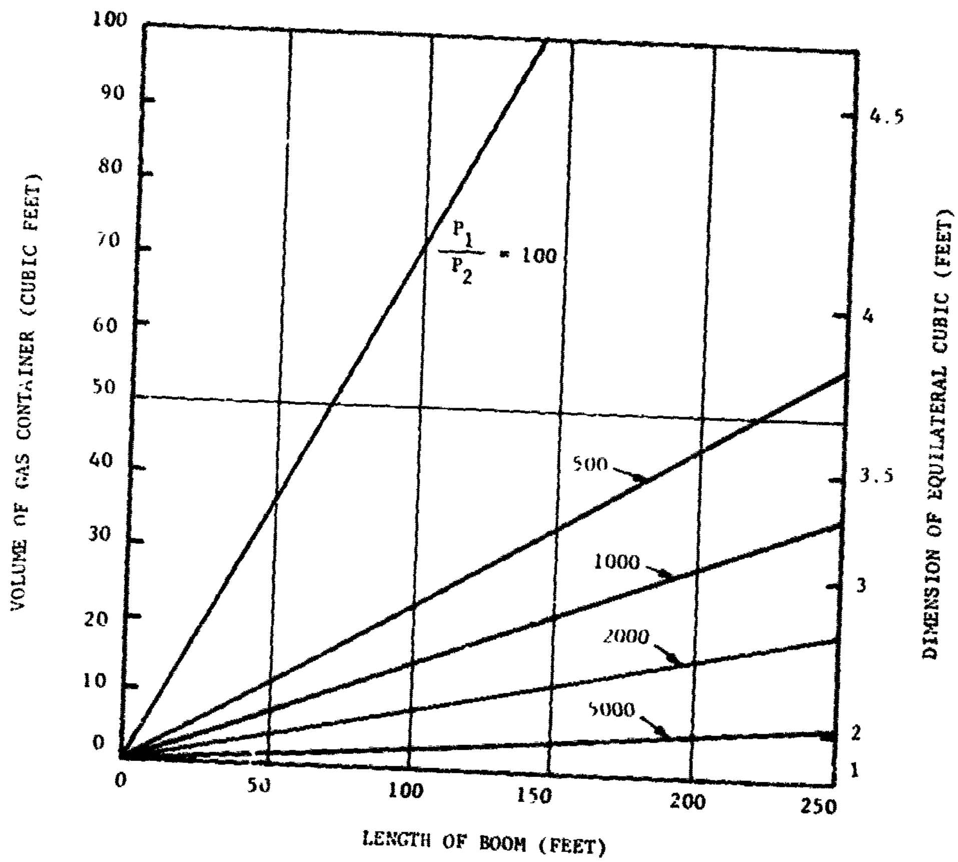


FIGURE 13. SCHEMATIC OF THE VARIABLE PRESSURE PNEUMATIC SYSTEM



BOOM DIAMETER: 5 FEET
P_1 = MAXIMUM PRESSURE IN CONTAINER
P_2 = NOMINAL PRESSURE IN BOOM
ASSUMPTION: ADIABATIC EXPANSION

R01879

FIGURE 14. VOLUME OF GAS CONTAINER VS. LENGTH OF BOOM

The function of the reduction valve is to control automatically the pressure within the structure. It provides:

(1) High initial pressure necessary to overcome initial static friction.

(2) Low pressure during deployment.

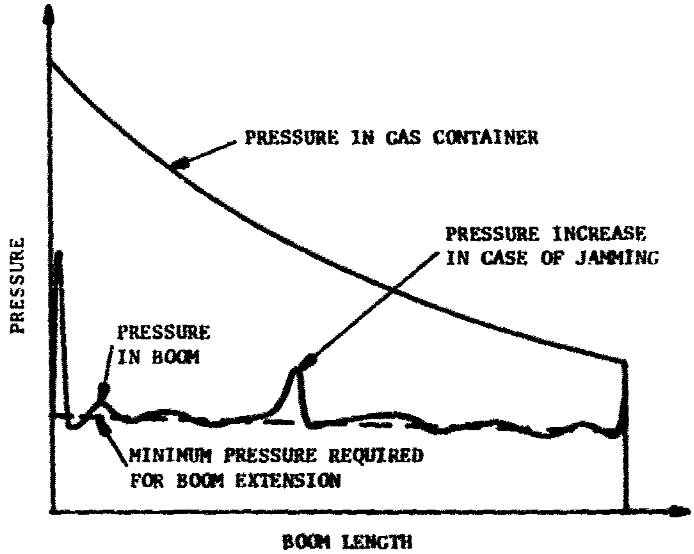
(3) Increased pressure in case of any unexpected jamming.

(4) Increased pressure at the end of the extension to eliminate any tendency of structural rebound.

When the squib valves open, pressure in the initially open portion of the boom rises to a level which is higher than the pressure necessary for extending the boom. Pressure drop in the reduction valve is small due to the small initial gas flow. The impact produced by the initial pressure pulse assures the separation. This is an important feature since static friction may be considerably higher than kinetic friction.

The pressures in the gas container and in the boom versus extended length of boom are shown in Figure 15. After the initial impact, which was discussed above, the boom starts to extend and the gas flow through the valve increases. Due to the increased gas flow and pressure drop in the reduction valve, the pressure in the boom drops to a level which is below that which is necessary for extending the tube. Consequently, the extension of the boom slows down and the air flow (and, the corresponding pressure drop) diminish. This, in turn, causes an increase in pressure, and the boom starts again extending. If the throttling is not excessively high, the resulting start-and-stop motion becomes a smooth motion.

An additional advantage of throttling is that the pressure in the boom is maintained at as low a level as is possible throughout the extension exercise. The associated unbalanced force, which may cause a deviation from the intended direction of the boom, therefore, is smallest. The optimum amount of throttling will have to be established experimentally.



R02027

FIGURE 15. PRESSURE IN GAS CONTAINER AND IN BOOM VS. LENGTH

THROTTLE REDUCES PRESSURE IN BOOM BELOW
MINIMUM PRESSURE REQUIRED FOR EXTENSION.

SUMMARY AND CONCLUSIONS

1. Four telescoping concepts were advanced which incorporate relatively elementary sliding movements adaptable to the mating sections of rigid construction.

2. One of these concepts was reduced to practice. This concept utilized a set of cylindrical metal sleeves and a coated-fabric diaphragm bellows. The coated-fabric diaphragm performed as a rolling seal between the metal sleeves. A series of friction tests on the seal indicated that the inherent friction in the system was very low.

3. Evaluation of seal integrity to pressure forces indicated no adverse leakage. This test was conducted at internal pressures of up to 15 psig, with results showing no measurable effect on the integrity of the fabric-metal bonded joints.

4. Evaluation of leakage rate associated with the demonstrated seal showed that approximately 200 minutes elapsed before the pressure dropped to one-half the initial level. The leakage test performed on the seal represented conditions that are much more severe than those that would actually exist in applications. That is, a static seal would ordinarily be employed at full extension; therefore, leakage as measured here would only occur during deployment.

5. The merits of the overall seal concept from the viewpoints of both mechanical movement and pressure integrity can be considered to be very good.

6. A materials investigation was conducted for obtaining a flexible diaphragm, proper bonding, and sealing. The materials that were found satisfactory consisted of a Dacron impregnated with a fluorinated elastomer and cement bases of fluorinated polymers, neoprene, and nitrile phenolic.

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2. Matacek, G. F., Vacuum Volatility of Organic Resins, WADC TR 59-268, September 1959. (Also, Matacek, G. F., Vacuum Volatility of Organic Coatings, paper presented at First Symposium, Surface Effects on Spacecraft Materials, Palo Alto, Calif., May 1959.)

G. T. Schjeidahl Company

Northfield, Minnesota

October 16, 1963

DESIGN AND FABRICATION OF INFLATABLE AND RIGIDIZABLE PASSIVE COMMUNICATION SATELLITES (ECHO I AND ECHO II)

Even today, with rockets available to carry heavy payloads into orbit, the thought of a 100-foot diameter orbiting sphere stimulates the imagination of scientists and laymen alike. At the inception of the Echo I program Sputnik I was a constant headline in all newspapers. Rocket payloads were limited to vehicles like the Vanguard, and this limitation gave added impetus to the field of inflatables or erectables for space. The field has been growing ever since.

Objectives of the Echo I setellite were many:

1. Demonstrate that a large inflatable device could be errected in space
2. Establish the feasibility of using a large reflective sphere in space for radio propagation
3. Measure radiation and solar pressure effects on orbit perturbation
4. Measure the density at orbit altitude
5. Obtain some idea of the life of plastic materials in space.

One objective, of course, was political. At a time when the U. S. Space accomplishments appeared to be lagging Russia by many years, we could give the world a satellite that everyone could see - a new star in the heavens.

DESIGN CONSIDERATIONS

In the design of the Echo I satellite, size was limited by the rocket (Thor-Delta) payload capability of approximately 200 pounds in a 1,000-mile orbit. This payload weight included the inflatable sphere, inflation material, canister, and separation gear.

ITEM	WEIGHT (POUNDS)
Sphere	120
Inflation Powder	30
Canister	40
Separation	10
Total	200

One-half mil metalized Mylar was chosen for the sphere material because it possessed the best balance of properties giving the largest possible reflector for the weight. Characteristics of this material include:

1. Availability - commercially available from du Pont in the thicknesses required.
2. Strength to Weight - Density of 1.39 and a tensile of 20,000 psi.
3. Flexibility - will withstand the flexing and folding required in packing and deployment in space without damage.
4. Bondability - sealed with GT*-301 to give seal strengths equal to the parent material under the temperature, radiation, and ultra high vacuum environment of space.
5. Radar Reflective - The vapor deposit of 2,500 angstroms of aluminum gives a surface which will reflect between 95 and 98 percent of the RF energy. (Equivalent to approximately 4 pounds of aluminum on the surface of a 100-foot diameter sphere).

* G. T. Schjeldahl Company Trademark Registered U. S. Patent Office

6. Radiation Resistance - The material was required to maintain its physical properties for a sufficient length of time to satisfy the requirements of the Echo experiment.

7. Thermal Balance - The temperature of the sphere when exposed to solar radiation must be sufficiently high to vaporize the inflating powders yet not exceed the limitations of the beacon or plastic.

8. Creep - Extremely low. Any creeping in the plastic will result in shape changes and breaking of the continuity of the reflecting surface.

In the early stages development, 0.25-mil metallized Mylar was used; however, through static inflation tests (Figure 1) and vacuum tank tests of rapid deployment from various shaped canisters, it was established that 0.5-mil Mylar would be required to give adequate reliability. The Echo I Satelloon* was fabricated of 78 gores bonded together with a 1-inch wide spliced plate of 0.5-mil Mylar bonded with GT-301 thermal setting adhesive system. These seams terminated in a 36-inch diameter polar cap at the top and bottom of the sphere. The cap was made of 1-mil metallized Mylar in order to carry the stresses transmitted to the polar cap area by the tape seams. The design provided a structure which could be stressed to between 15,000 and 18,000 psi at room temperature, 12,000 psi at -70 C, and 5,000 psi at a temperature of 100 C. Two wafer-thin beacon transmitters powered by solar cells were located 180 degrees apart on the sphere surface.

* G. T. Schjeldahl Company Trademark Registered U. S. Patent Office

STATIC INFLATION TEST ECHO I

100' diameter sphere of .0005 inch thick metallized Mylar and sealed with Schjel-Bond 201.

Figure 1

During fabrication, 10 pounds of benzoic acid and 20 pounds of anthraquinone subliming powders (which change to a gas in the near vacuum of space) were dusted on the inside of the surface of the deflated sphere. The sphere was then accordion folded and placed inside the 26.5-inch diameter magnesium container (see Figures 2 and 3). About two minutes after the payload was ejected into orbit the magnesium container was separated by an explosive charge. This linear-shaped charge served to cut the stays holding the canister halves together and at the same time separated the canister halves at a velocity of 40 fps* . The 10 pounds of benzoic acid then expanded to inflate the Echo into a spherical shape in a matter of seconds. The theoretical skin stress developed by the benzoic acid is 100 psi. The anthraquinone having a higher vapor pressure is used to provide a progressive sustaining inflation material.

Extensive vacuum tank tests and vertical ballistic rocket shots were used to qualify the balloon payload. In the initial phases of development, water contained in small plastic bladders and ejected through a controlled orifice was used to inflate the spheres. The first test of a 100-foot diameter sphere using water inflation was not successful, and the sphere burst into many fragments. By launching these balloons east into the moving sunrise and using an indicator dye within the sphere it was possible to observe from the ground whether or not the balloons burst.

* Canister tests in the 40-foot vacuum chamber at NASA Langley has demonstrated that this velocity is sufficient to move the canister halves away from the rapidly inflating sphere.

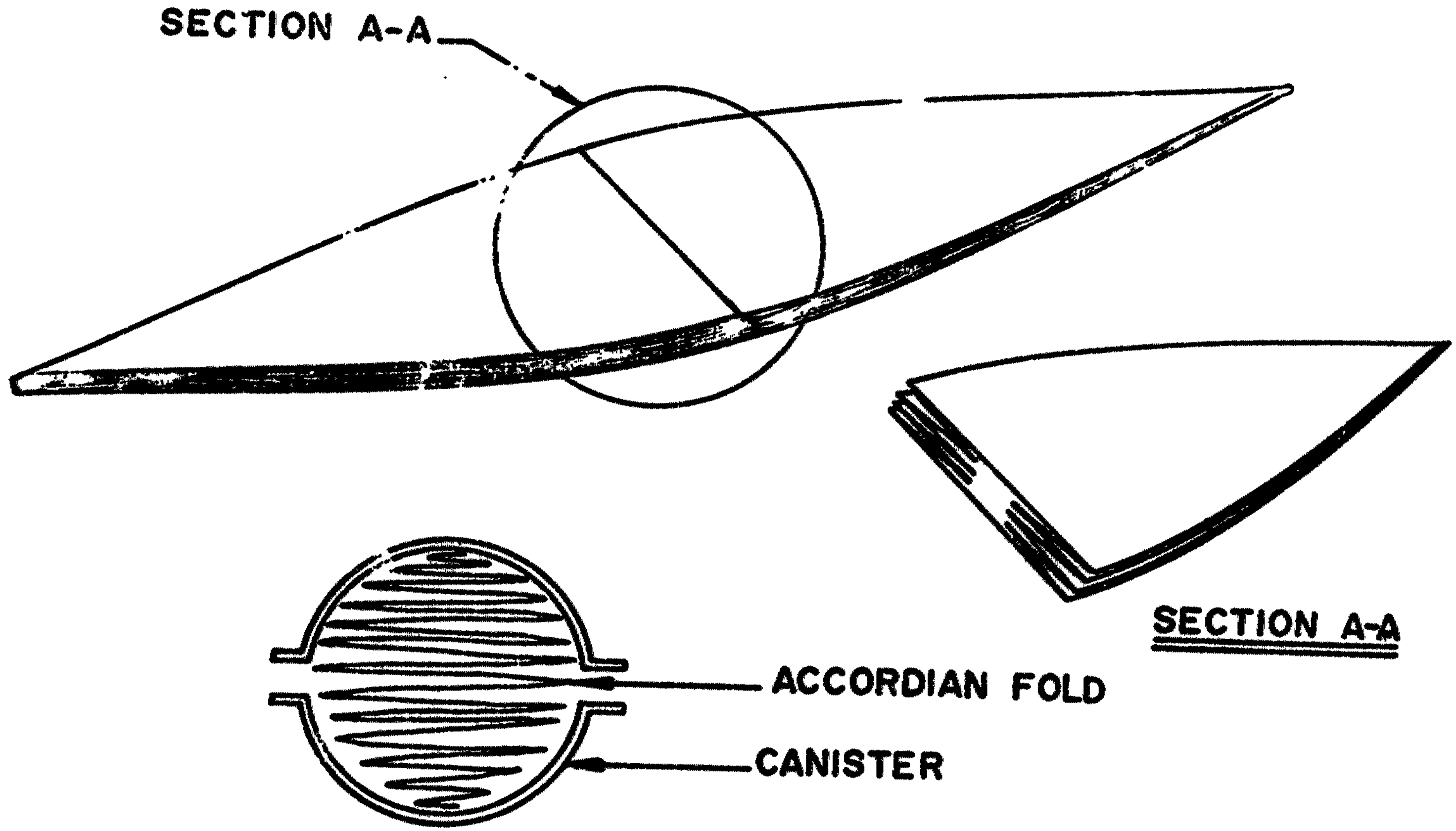


FIG. 2

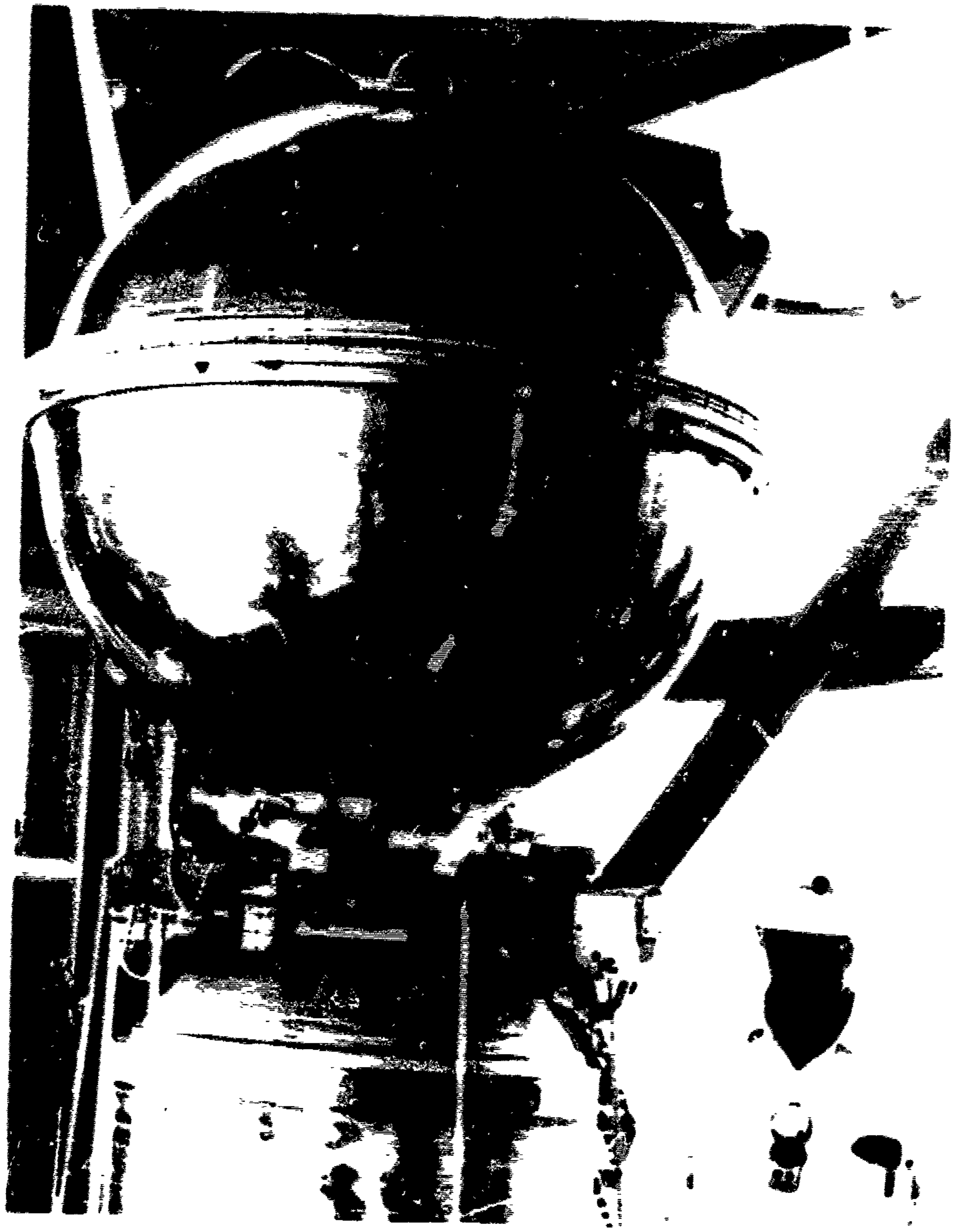


Figure 3

In the second test it was decided to employ a less volatile inflation material, i.e. benzoic acid, and a pressure within the balloon container of 140 millimeters of mercury. It was felt that this lower pressure would slow down the initial deployment resulting from the entrapped air and the subsequent inflation from the less volatile subliming material. Although this sphere did not show the explosive characteristics of the first test, it too ruptured into several pieces. A third test was then scheduled reducing the pressure inside the canister to 50 millimeters of mercury. This showed further improvement over the second test with only one rupture expelling some of the red dye. In the fourth and fifth tests the pressure inside the balloon canister was reduced to 4.5 millimeters of mercury prior to launch. Both of these tests were successful with no signs of balloon bursting or red dye in the vicinity of the inflated balloon. With these two successes the design was frozen and work proceeded on the orbital launch of Echo I.

At the time Echo I was placed into orbit, it was calculated that it would stay spherical for approximately two weeks. This was based on pre-orbit calculations placing the satellite in continuous sunlight for a two week period and the fact that micrometeorite punctures would eventually release the subliming materials.

Based on available data, it appears that the Echo satellite performed as predicted giving a fairly uniform, spherical shape for at least two weeks. Radio and television signals have been bounced from the Echo satellite, thus demonstrating its feasibility for communications. The success of Echo I has gone far beyond the most optimistic expectation. Today, more than three years after its ejection into orbit, it is still serving as a research tool for communication and space scientists. The quality of the signals being received from Echo, however, have gone through a considerable change, indicating that the sphere has changed its shape or possibly become wrinkled. Its appearance from the ground appears unchanged from that observed when it was first placed into orbit.

Echo I was not designed to be rigid enough to withstand the deforming forces of space, e.g. a solar pressure of 1.3×10^{-9} psi. Changes in shape were, therefore, expected.

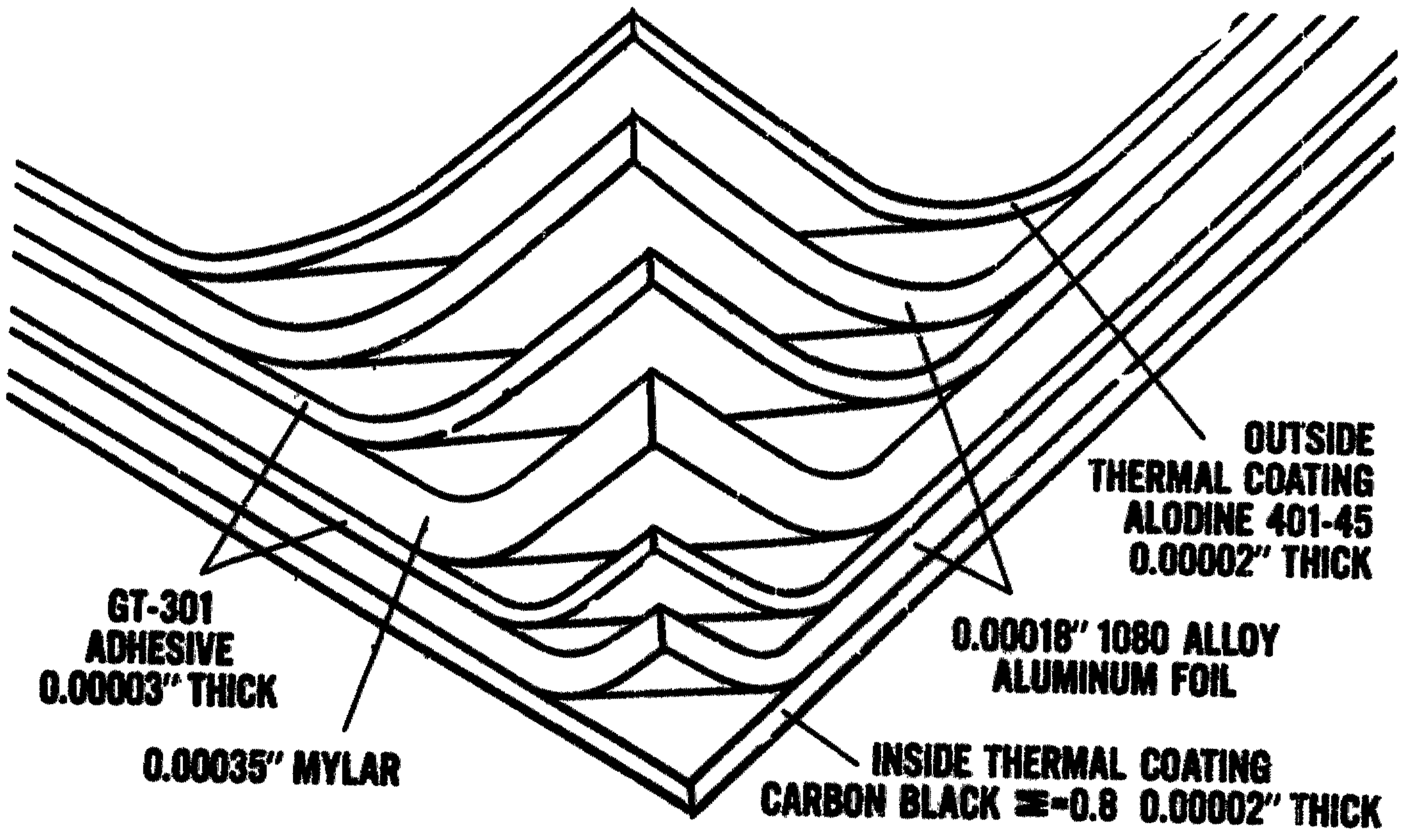
ECHO II

As a result of the success of Echo I, NASA undertook the development of a more permanent type satellite, one which would be rigid enough to maintain its shape when exposed to space environment. The goal, as in the case of Echo I, was to build as large a sphere as possible within the weight limitations of the vehicle, in this case the Thor-Agena B. Basic material requirements for the Echo II Satelloon were the same as Echo I with the exception of the ability to be rigidized in space.

Several approaches for accomplishing the rigidization of Echo II were considered. These included such concepts as mechanical stressing of a metal-plastic laminate, chemical cross linking, electrostatic repulsion, and plasticizer evaporation from a rigid plastic film. Although the chemical and volatile plasticizer concepts yielded some materials with interesting properties, the simplicity, reliability, and lightweight of the mechanical approached to rigidization resulted in the choice of an aluminum-plastic laminate for the Echo II satellite.

Through an extensive materials design and development program, a seven-layer material 0.75-mil thick and weighing 36 gm/sq. yd. was manufactured in a 54-inch width to meet the requirements of the Echo II program (see Figure 4). This material is basically a lamination of 0.00018-inch thick aluminum foil on both sides of 0.00035-inch thick Mylar. In order to control the inflation material (pyrazole) and keep the temperature of the sphere within that which the acquisition beacons would operate, it is necessary to thermal balance treat the inside and outside surface of this material. Alodine 401-45*, a chemical-conversion coating for aluminum, is used on the outside surface to change the ratio of solar absorptence to emittance from 6.5 to 1.7. The inside surface of the sphere is coated with a carbon black to increase the heat transfer from one side of the sphere thereby helping to control the temperature gradients and reduce hot spots.

* American Chemical Trademark Registered U. S. Patent Office



ECHO II LAMINATION

FIGURE 4

The Echo II sphere will be rigidized by stressing the dead soft aluminum foil by means of an internal pressure provided by 38 pounds of pyrazole subliming compound packed in the sphere. When the folds in the material have been removed by the pressurizing of the sphere the high modulus of elasticity of the aluminum foil will resist the deforming loads so that the structure will retain its spherical shape without internal pressure. The Echo II sphere is computed to have a rigidity of 50 times that of Echo I.

The Echo II sphere is 135-feet in diameter, having a cross-sectional area of two times that of Echo I. It is fabricated in approximately the same manner as Echo I, with 106 tailored gores approximately 48 inches wide sealed together with a 1-inch wide splice plate of the same material as used in the balloon gore (see Figure 5). The adhesive system is GT-301 thermo-setting system developing a seam strength equal to the material strength over a temperature range from -200 to 150 C. These seams are further tested by flexing and folding or cycling between liquid nitrogen and boiling water. The end caps of the sphere and the radio beacon installation areas (180 degrees apart on the equator of the sphere) are reinforced to carry the added material load (see Figure 6). One-hundred and sixty air orifice holes 1/64-inch in diameter are placed in the top and bottom layers of the folded balloon to permit removal of air, moisture, and volatile impurities from the inside of the sphere during canister pump down.

Figure 5

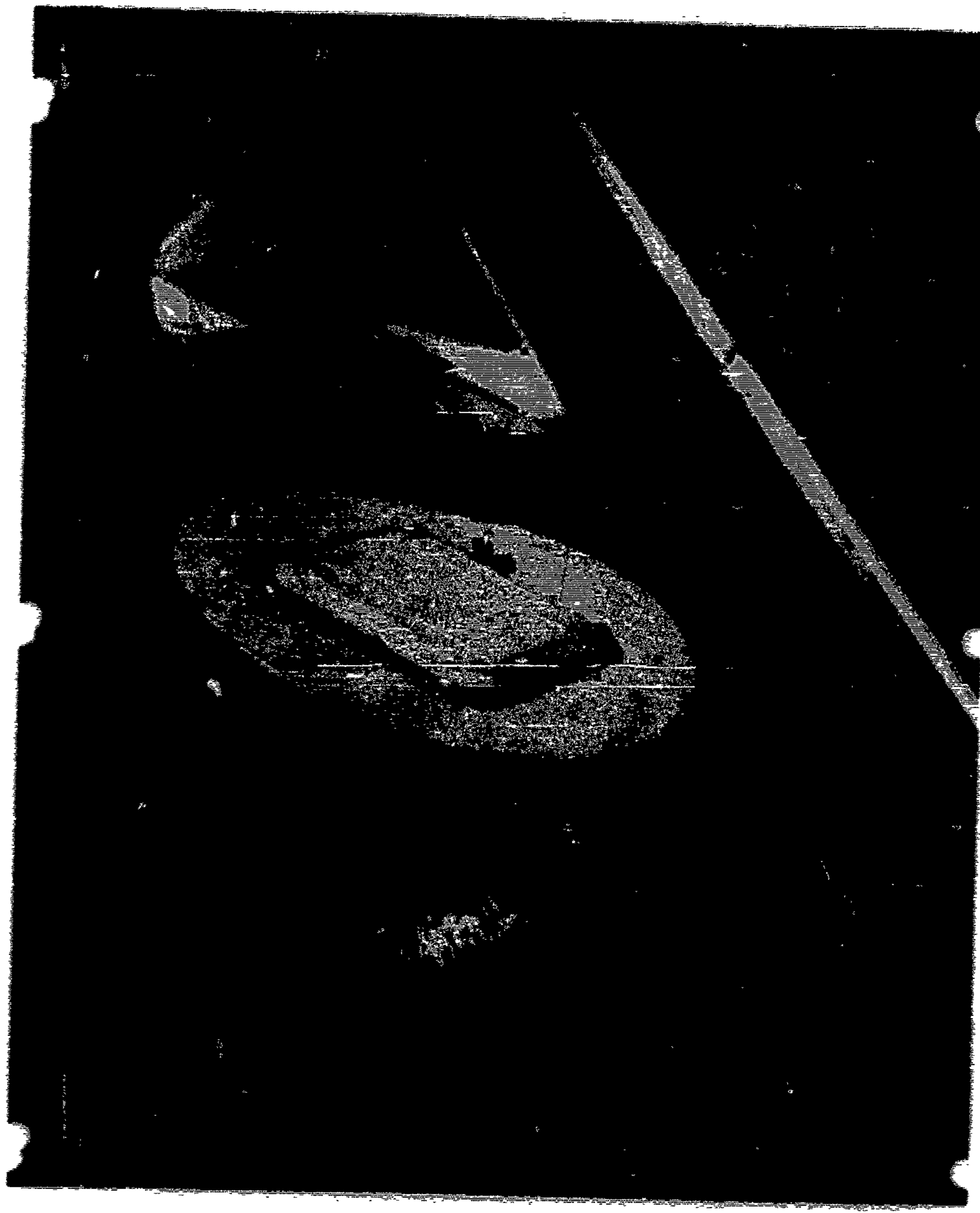


Figure 6, Transmitter and Solar Modules

Figure 7 shows a balloon and canister being placed in the vacuum chamber prior to evacuation. The sphere canister halves are separated by 1/8-inch during the evacuation. After a 40-hour soak at a pressure of 200 microns the canister halves are closed and sealed by means of electrical wrenches operated from outside the vacuum chamber. The total weight of the Echo II balloon system is broken down as follows:

ITEM	WEIGHT (POUNDS)
Balloon Weight	469
Sublimination Material	38
Thermal Balance Coatings	33
Beacons	12.6
Canister	70
Total System Weight	622.6

THERMAL BALANCE

The thermal balance coatings on Echo II was a significant development and merits further discussion here. The thermal balance on Echo I was achieved through the high reflectivity of the vapor deposited aluminum on the outside of the sphere and the high emissivity in the infrared of the Mylar on the inside of the sphere. Echo I was calculated to be 110 to 134 C.

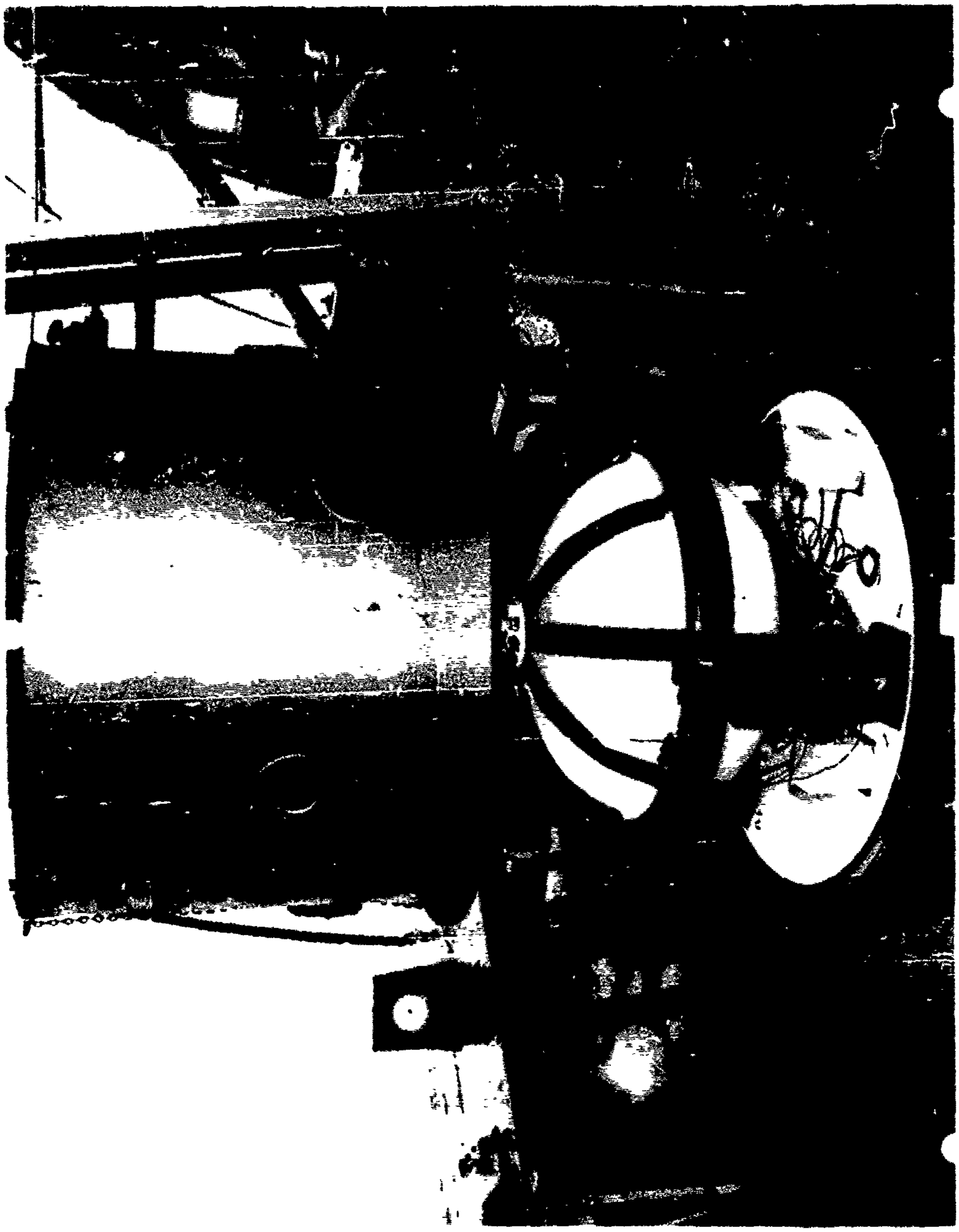


Figure 7

If the aluminum foil used on Echo II is left untreated, the average temperature has been calculated to be in the range of 181 to 210 C*.

The temperature control of the Echo II satellite was very important for proper operation of the sublimination material in deploying and rigidizing the sphere, and in operation of the acquisition beacons which would not function properly above 60 C.

A number of materials were investigated to reduce the temperature of the sphere; however, in most cases their weight was equal to or greater than the basic material itself. Most inorganic compounds associated with inks and paints are adversely affected by extended exposure to ultra-violet and ionizing radiation and it was therefore decided to focus investigation on areas of inorganic thermal coatings having a greater stability. The results of this study was a commercially available material called Alodine 401-45 which offered the greatest change in α/ε ratio for a minimum of weight. This coating is used commercially as a primer for painting aluminum or for corrosion resistance of aluminum. Clemmons and Camp of NASA have determined the change in α/ε ratio as a function of the surface density or coating weight for Alodine 401-45 (see Figure 8). From this study it was determined that an α/ε ratio of 1.7 results in an average temperature of about 55 C on the satellite.

* Clemmons, Dewey L., Jr., and Camp, John D.,"Amorphous Phosphate Coating for Thermal Control of Echo II." Paper presented at the Multi-Layer Systems Symposium of the Electro-Chemical Society Meeting, Los Angeles, California, May 6, 1962.

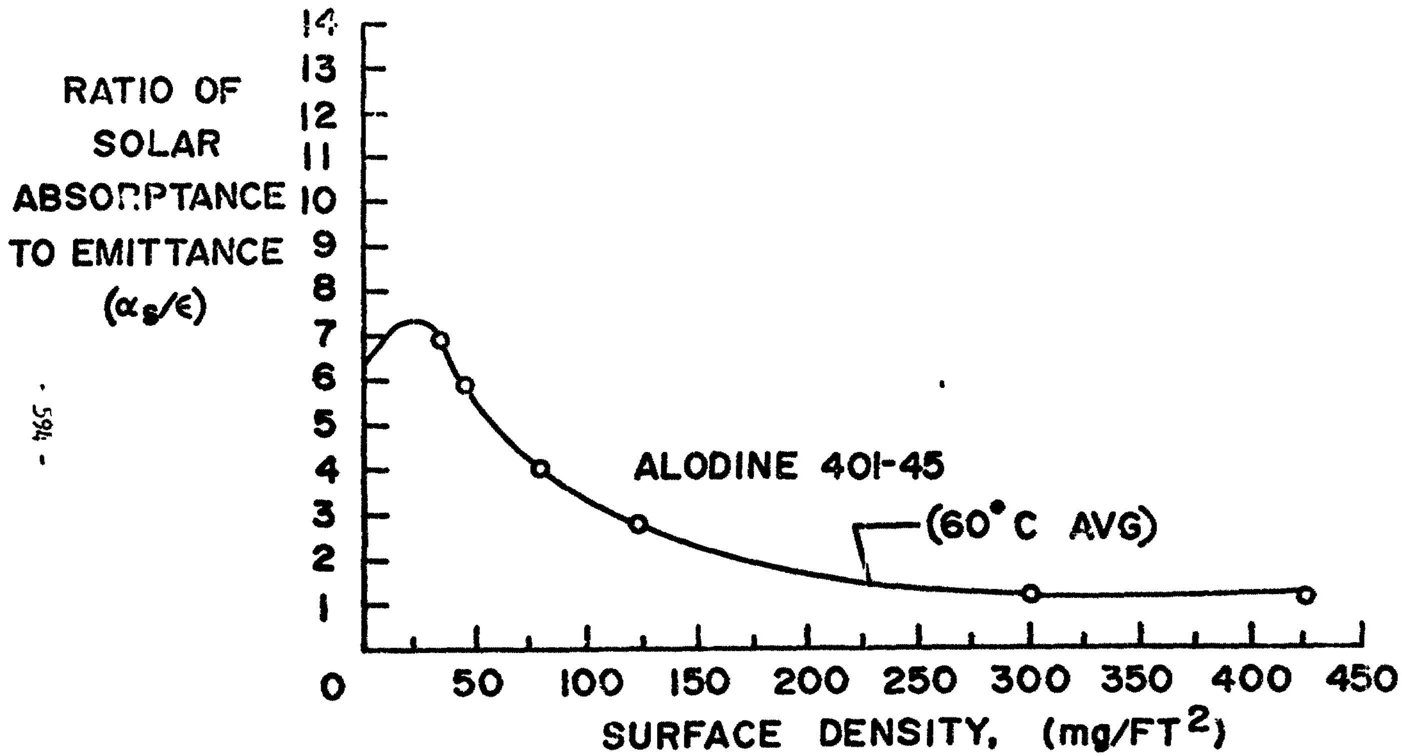


FIG. 8 VARIATION OF α_s/ϵ RATIO WITH SURFACE DENSITY.

The variation of solar absorptance and the emittance for Alodine 401-45 as a function of surface density is shown in Figure 9. The solar absorptance remains essentially constant from 50 through 300 milligrams per square foot, whereas the emittance varies almost linearly with thickness.

The finding of a suitable thermal balance coating was only part of the problem involved in thermal balance treating of Echo II. There was no known method for applying Alodine to a 54-inch web thousands of feet long and maintaining a tolerance of $\pm$ 3 milligrams per square foot. The Schjeldahl Company, with experience in the plastic machinery, lamination machinery and printing machinery manufacturing fields, undertook for NASA the design and construction of equipment to handle this task, (see Figure 10). Although the yields from this process were far from commercial standards, material for the manufacture of Echo II satellites with an Alodine coating tolerance of $\pm$ 3 milligrams per square foot was achieved.

TESTING

As in the case of Echo I, extensive vacuum chamber and ballistic trajectory tests were required in order to qualify the payload system. Figure 11 shows one of the first Echo II satellites inflated in the hangar at Weeksville, North Carolina. This satellite ruptured at a skin stress equivalent to 18,000 psi on the aluminum. This was considered an adequate safety factor

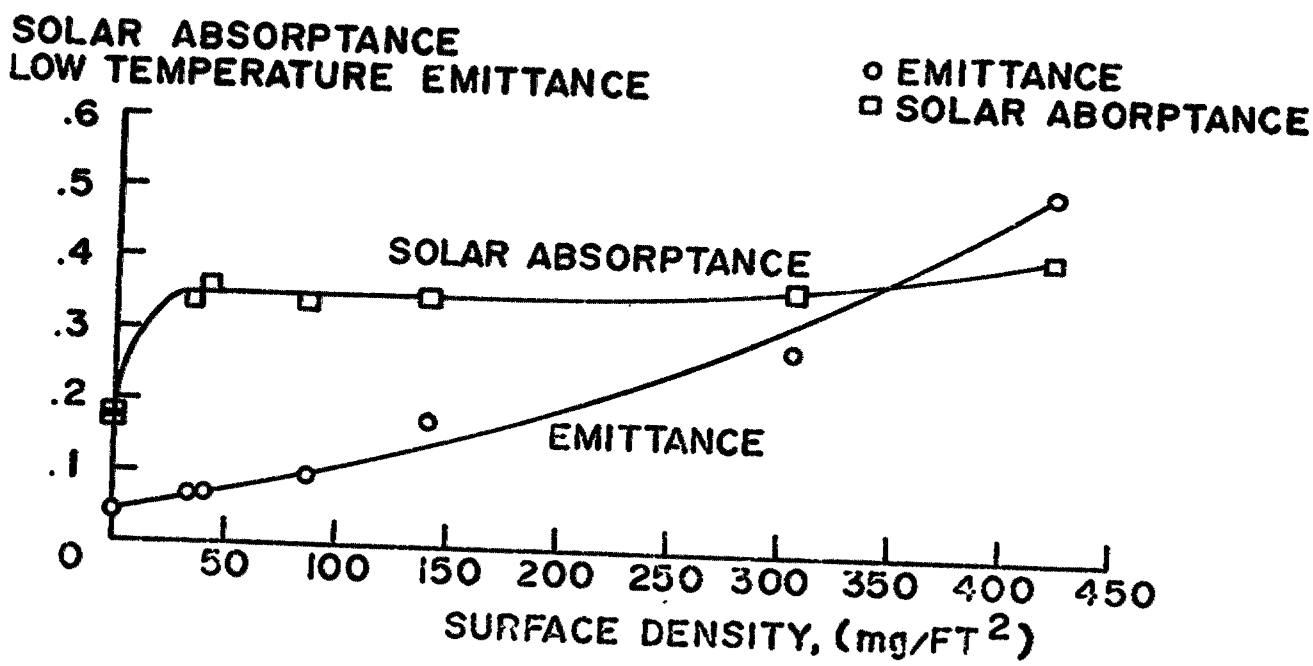


FIG. 9 - OPTICAL PROPERTIES OF ALODINE.

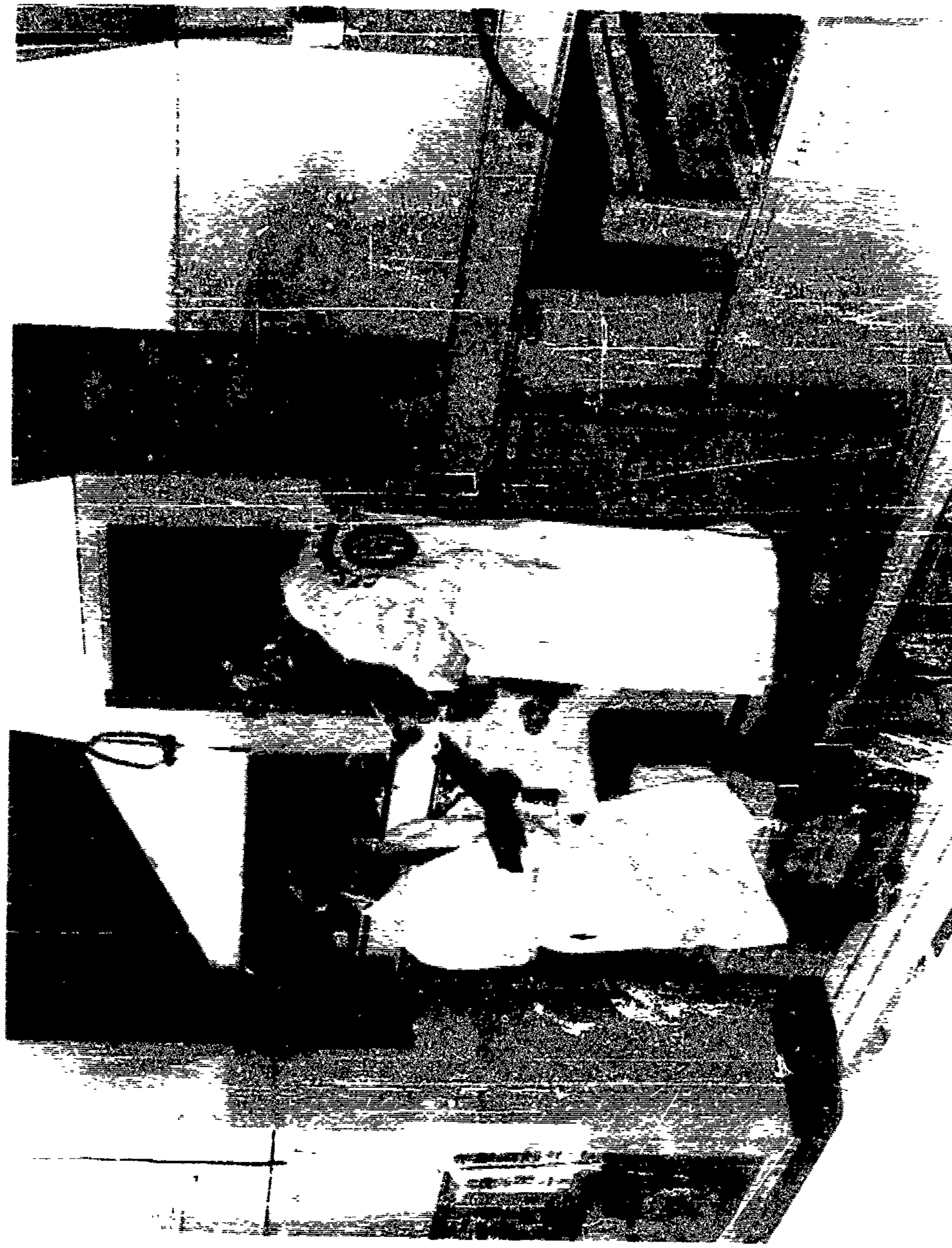
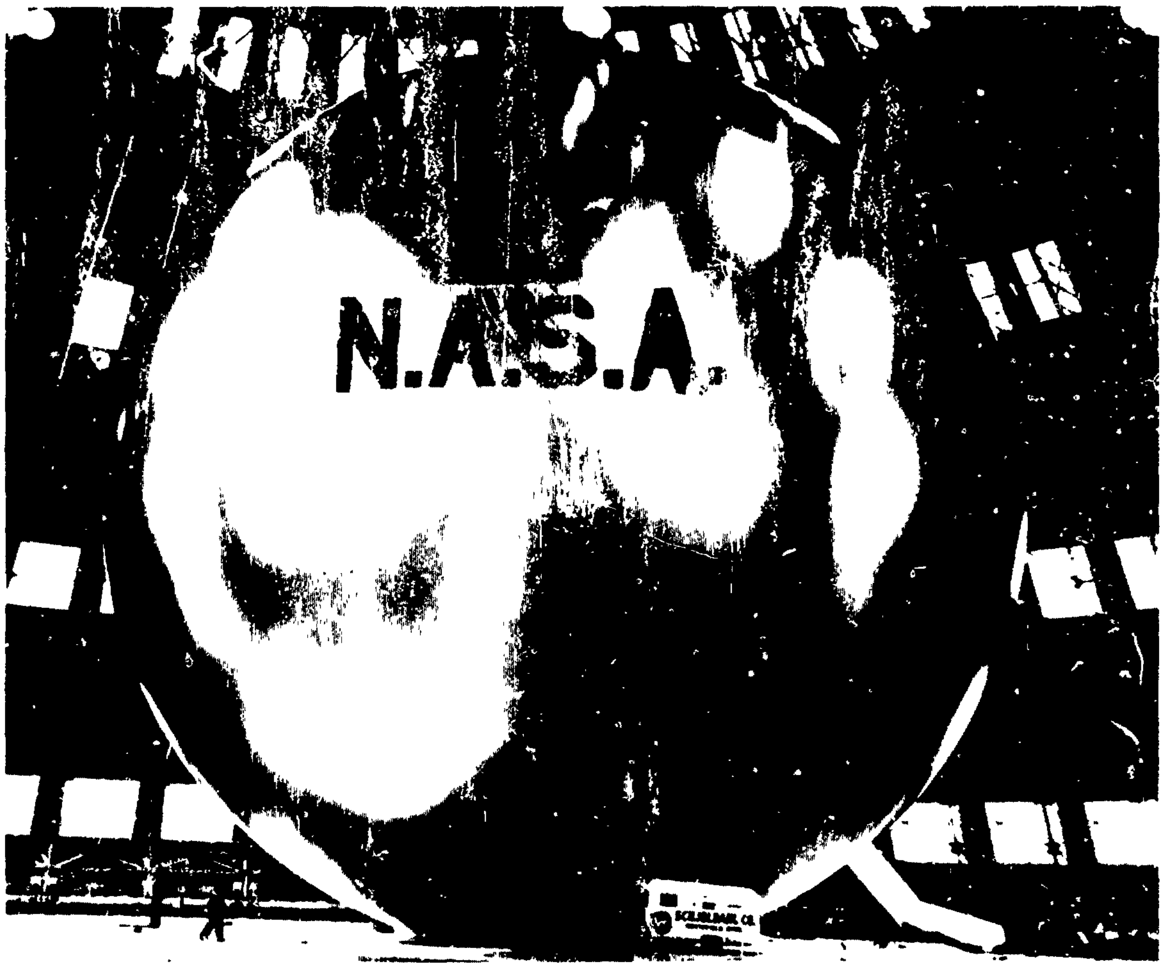


Figure 10



STATIC INFLATION TEST ECHO II

135' diameter sphere built of .00018" thick aluminum foil on both sides of .00035" thick Mylar bonded with 301 'Schjel-Bond'

considering that 3,000 psi skin stress would be required to remove the fold memory and rigidize the satellite.

The 61-foot diameter vacuum chamber at NASA Langley was used to determine the separation velocity of the canister and the initial deployment of the sphere. It was recognized that these tests were only indications of how the satellite would deploy in space. The sphere fell rapidly to the bottom of the chamber, restricting further deployment, and thermal and pressure conditions of space could not be achieved. Therefore, to test the complete system under actual conditions a vertical ballistic shot was scheduled to provide a no gravity, ultravacuum environment for a period of approximately 20 minutes. In the first vertical test for Echo II on January 15, 1962, the inflation process was recorded by a recoverable motion picture camera mounted in the front end of the Thor Rocket. A television camera which relayed back the real time pictures of the inflation during the test was also mounted on the Thor. This test revealed several serious problems which, when combined, cause the sphere to rupture during an explosive inflation in less than two seconds. A considerable amount of information was gained from these flights and applied to a redesign for a second AVT test. These include:

1. The residual air pressure in the canister was reduced prior to launch from 10 millimeters of mercury to 1 millimeter of mercury in order to reduce the initial deployment speed.

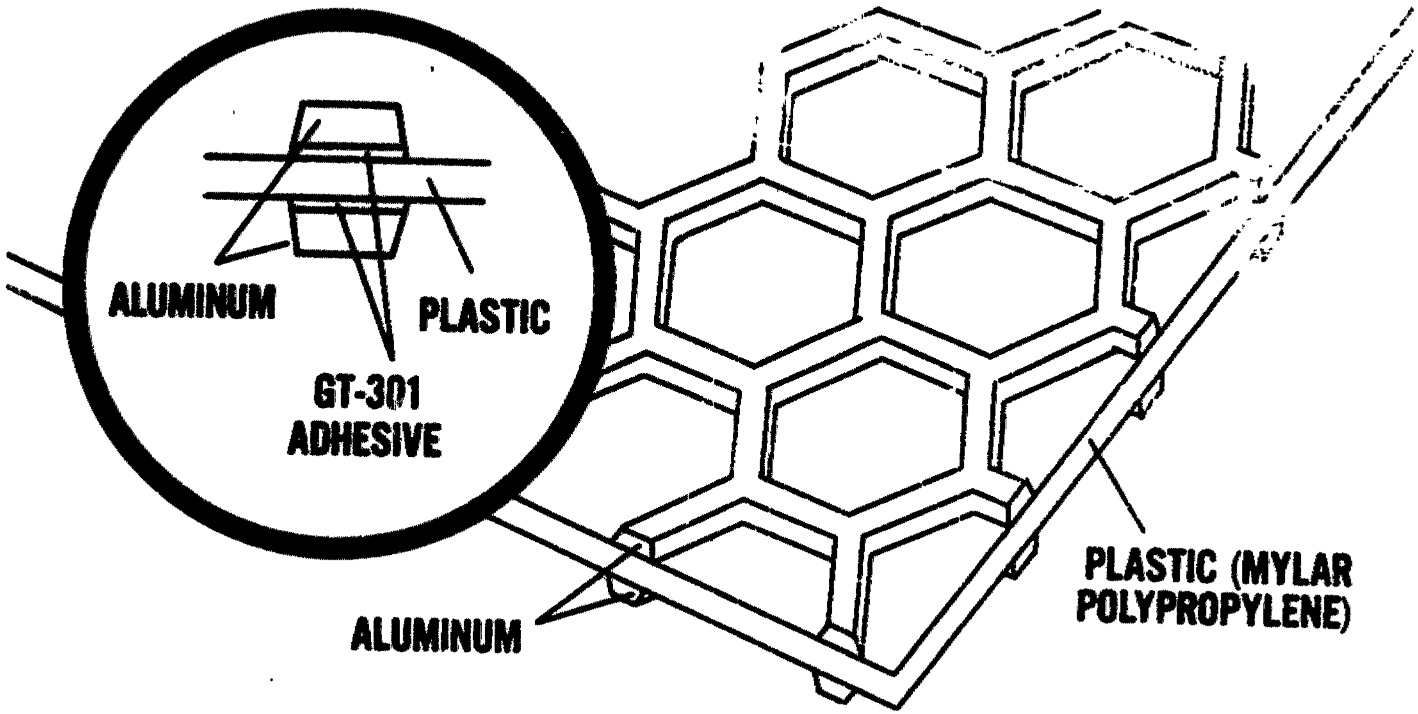
2. A less volatile sublimination material, i.e. benzoic acid was used to reduce the inflation and rigidizing process.
3. The sphere within the canister was evacuated to 200 microns and held for 40 hours for complete removal of water vapor and other impurities.
4. The beacon gores and areas around the beacons were reinforced.
5. The folding pattern was modified to slow the inflation of the sphere with respect to the equator.

On July 18, 1962, a second vertical test shot was launched from Cape Canaveral Florida. Television and camera coverage of this test showed a significant reduction in the speed of deployment. This sphere was fully deployed in 10 seconds and appeared to be rigidized within 30 seconds.

Although this test appeared successful from the standpoint of balloon deployment, radar signals bounced from this balloon showed scintallation which exceeded that expected for this satellite. This was believed to be due to the less volatile sublimination compound (benzoic acid) which theoretically pressurized the sphere to only 100 psi skin stress. The problem then was to design the inflation system around a material which 1) give adequate inflation pressure to rigidize the sphere and 2) be controlled in such a way that it would

not result in explosive deployment and inflation. The result of this study was a system incorporating pyrazole as an inflation material packed into 72 wax sealed plastic bags. These bags were attached to the surface of the sphere at locations which would give even deployment and inflation. The wax used to seal these containers was designed to melt at 37 C allowing the pyrazole to escape through a number of small holes. Preliminary tests conducted on this inflation system looked very promising and work is proceeding on the orbital balloons to be shot into orbit early next year.

Concurrent with the work on Echo II materials and designs aimed at reducing the weight of a given passive communication reflector is under way. One of the new materials being investigated is a lamination of plastic and aluminum (Figure 12) similar to that used on Echo II. This material employs a biaxially oriented polypropylene with a tensile strength of 30,000 psi in both directions and a density of 0.92. Polypropylene was chosen because of its high strength-to-weight ratio and its resistance to discoloration in a space environment. The aluminum in this case is milled out in a hexagon pattern to 1) reduce the satellite weight and 2) allow 90 percent of the solar energy to pass through the satellite thereby reducing the required rigidity.



ADVANCED MATERIAL FOR RIGIDIZED COMMUNICATION SATELLITES

FIGURE 12

Further study to reduce the weight of these communication satellites is presently under study. One technique employs a lenticular shape reflector which is oriented by means of gravitation gradients (Figure 13).

The simplicity, service life, and anti-jamming characteristics of passive communication satellites make them a very reliable communication tool. Through new materials and design, the efficiency of these inflatable satellites (more surface area for less weight) is increasing.

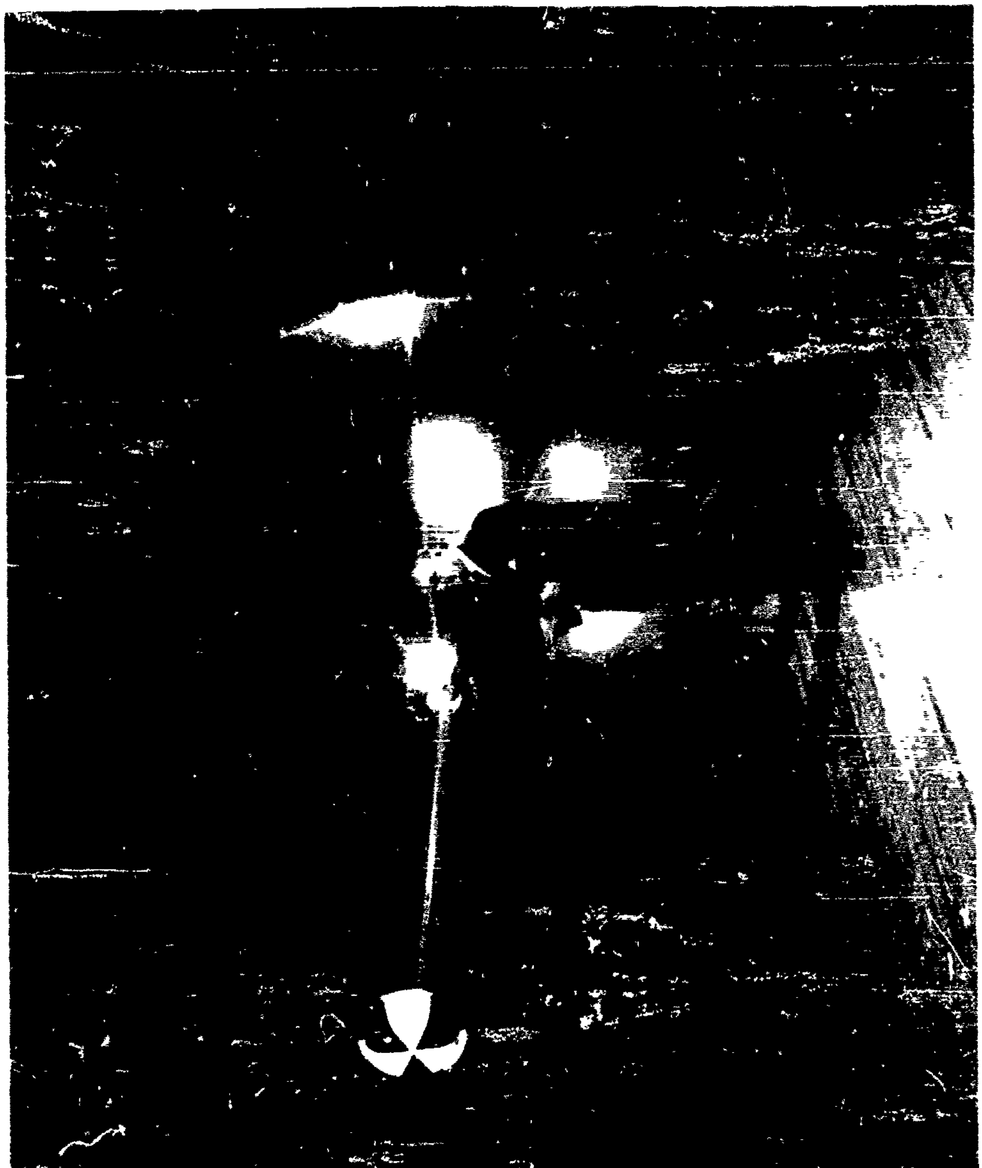


Figure 13

INFLATABLE SPACE ANTENNA SYSTEM

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ABSTRACT

A recent development of BIRDAIR Structures, Inc. has been the design, development, and fabrication of a series of active inflatable space antenna systems. Each antenna system, which is comparable in size to a compact automobile, is an assembly of complex inflatable shapes arranged and held to extremely close dimensional limitations. Some of the major problems involved in the development of the antenna system, from a mechanical view point, were the development of materials suitable for inflation and rigidization in a space environment, development of a container and pressurization system, providing required RF performance and reliability with minimum weight, and the development of folding and deployment techniques for a relatively complex inflatable configuration.

INTRODUCTION

Since certain design features of the antenna system and certain performance characteristics are classified, the discussions given in this paper are necessarily limited to the unclassified aspects of the program and can only describe general accomplishments in this new field.

The success of the Echo balloon satellites, which have been placed in earth orbit, and plans for future satellites of this type are indicative of the feasibility and inherent advantage of using inflatable devices for large size, minimum weight, space structures. As of this date, however, such devices have been primarily in the category of inactive or passive satellites. BIRDAIR Structures has recently developed an active inflatable antenna system consisting of a group of separate antenna assemblies held together by an inflatable connecting structure. Experience gained in the development and fabrication of numerous inflatable parabolic antennas and solar reflecting mirrors, all held to relatively close tolerance, provided BIRDAIR with the background necessary to undertake this program. Based on antenna configurations which provide the desired electronic performance, BIRDAIR undertook to provide the mechanical components of the antenna system in the form of an inflated structure which would satisfy these electronic requirements.

The major criteria considered in the development of the inflatable antenna system were:

1. to provide the desired complex electronic configuration in an inflated form having the lightest possible weight.
2. to fabricate the inflated structure to extremely close dimensional accuracy.
3. to provide the smallest possible package size for installation in the launch vehicle.
4. to be self rigidizing after initial inflation while in earth orbit.
5. to provide the desired electronic configuration in an inflatable shape.
6. to survive space environment for a substantial period of time.
7. to provide a cost advantage over a comparable rigid mechanical antenna system.
8. to provide the pressurization system and container for carrying the antenna into orbit and subsequently deploying it into its final configuration.

GENERAL DESCRIPTION

The complete inflatable antenna system consists of three basic components: the inflatable antenna envelope, the pressurization system, and the container for carrying it into orbit.

Each of the antenna elements and the connecting structure consist of parts and segments of spheres, cylinders, cones, tori, oblate spheroids, and flat tensioned sheets. All elements of the complete structure are tensioned into final shape by the inflation pressure.

In preparation for launch the antenna system will be initially evacuated and folded inside an evacuated canister. At an appropriate time after the transport vehicle has been placed in orbit, an extension mechanism to which the canister is affixed will extend outward from the vehicle to the outboard position. Upon full extension, the canister cover will be ejected clear of the folded antenna. During a time interval of 30 seconds after canister ejection, a pair of spring-loaded arms will commence unfurling the antenna system. At the end of the 30-second interval, nitrogen gas will be released into the antenna, inflating it to the approximate final shape in 30 seconds, and reaching its peak pressure in approximately 30 minutes. The pressure will then bleed down to ambient pressure. After the pressure differential has diminished, the antenna system will retain its shape and dimensional stability in the zero gravity condition.

ANTENNA ENVELOPE

The complete antenna envelope when inflated measures approximately 72 inches in height, 84 inches in width, and 72 inches in depth, yet weighs under 8 lbs. The inflated volume of the antenna system is approximately 106 cubic feet. This complex antenna system after evacuation and special folding is contained in a package measuring 10 inches x 10 inches x 7 inches. Each of the antenna elements, although not parabolic reflectors, consist of envelope, reflector, and feed assembly. Each antenna element was fabricated within overall tolerances of $\pm$ 1/16 inch. The group of antenna elements, when connected by the central support structure, were maintained in proper relationship within a tolerance of $\pm$ 1/16 inch.

In the development of the inflated structure, it became evident that both metallic and non-metallic materials would be required in order to meet the mechanical and electronic requirements. Since it would not be possible to maintain inflation pressure in the structure for the entire life of the antenna system, it was necessary that the antenna material be self rigidizing after the initial inflation. After a thorough review of materials suitable for use in a space environment, several combinations of plain Mylar and laminates of Mylar and aluminum foil in various gauges were chosen for the envelope material as listed in Table 1. With the selection of the envelope material, attention was then given to the selection of a suitable adhesive for bonding the Mylar and foil panels together. An adhesive capable of carrying inflation loads at both the high and low temperature extremes encountered in a space environment, capable of sustaining radiation as encountered in a space environment without degradation, and possessing a minimum outgassing characteristic was required.

Both the envelope materials and the adhesive selected for the antenna envelope were then given a series of stringent material tests in order to prove their suitability for this application. The testing program conducted with these materials consisted of tests for basic material strength, effect of temperature, gas holding properties, effect of repeated folding, inherent rigidizing properties, effect of extremely low pressures (vacuum conditions), and electrical conductivity.

TABLE 1
Properties of Antenna Materials

Material	Thickness Inches	Strip Tensile Strength Lbs. Per Inch	Weight Oz. Per Sq. Yd.
1. Polyester film (Mylar)	.0005	7	0.52
2. Aluminum foil/Mylar/aluminum foil	.00035/.00025/.00035	9	1.98
3. Aluminum foil/Mylar/aluminum foil	.0005/.0005/.0005	18	2.9

The aluminum foil and Mylar laminates were found to have excellent rigidizing characteristics for relatively thin-film materials. One of the suggested methods of rigidizing inflatable structures fabricated from an aluminum foil and Mylar laminate is to tension the foil beyond its yield point. This approach was investigated but was considered impractical for this application. Because of the irregular shape of the antenna, the stress under uniform inflation pressure would vary appreciably from point to point. While it is possible to vary the gauge of foil used in the different components, the varying stresses would make it impossible to provide uniform stress conditions. After reviewing these factors and experimenting with various aluminum foil and Mylar combinations, it was decided to take maximum advantage of the inherently stable shapes of the antenna components and the possibility of readily shaping the very ductile foil by means of pressure. Since the plain Mylar film was only used for closing membranes to make the antenna gas tight and not for any actual electronic function, possible distortion of this material after deflation was not considered significant.

The materials and combinations of materials used in the fabrication of the antenna were exposed from four to twelve hours in a vacuum of from 50 to 100 microns. No change was noted in any of the materials following this exposure.

Fabrication of the antenna system to the extremely close tolerances required presented several problems in the development of assembly techniques. Considerable effort was spent in the study of patterning the flat materials in order to facilitate fabrication, yet maintain the proper final shapes. Both the method of fabricating the antennas as free envelopes by assembling panel to panel (as is done with large inflatable structures) and the method of assembling the panels on a form or mandrel were considered. Although extreme care was required in the assembly of joints in the antenna components, the free-envelope method of construction was selected due to the complex shapes required. Consideration was given to the problem involved in maintaining electrical continuity between the various components of the antenna system, including the use of conductive cements. However, it was determined that satisfactory electrical performance could be obtained through capacitive coupling of the lap joints.

In conjunction with the fabrication of the antenna, it was also necessary to develop and fabricate a flexible feed system for each of the three antennas which could be packaged and folded for deployment with the antenna as it is inflated. Power was carried to the antenna elements through flexible RG138 coaxial cable. Routing of the flexible coaxial cable from the attachment point on the launch vehicle to the three antenna feed systems so as not to become tangled during inflation and not to distort the antenna after deflation, was also a problem to be considered.

As with previous inflatable satellite systems, temperature control of the envelope material was considered extremely important. It was determined from earlier studies of inflatable earth satellites that uniform envelope temperatures can be approached and moderate temperature limits can be achieved if the emissivity ratio between the inner surface and the outer surface is kept high, and if the absorptivity to outer surface emissivity ratio is kept low. Two sun-earth-satellite configurations were studied to

determine temperature extremes. These are (1) when the satellite is in the earth's shadow, and (2) when the satellite is between the earth and the sun. After a study of the factors involved in temperature control, it was found that temperatures of the antenna envelope could be controlled within a range of -125° F. to +200° F. by treatment of the outer surface of the antenna to obtain the proper absorptivity to emissivity ratio. The temperatuve control treatments for the antenna system consisted of coating the inside of all enclosed components with flat black acrylic paint, treating the outside of the antenna components with 401-45 alodine to a coating weight of 200 mg. per square foot and painting the exterior of the antenna envelope with 2 inch diameter "polka dots" of oyster white acrylic paint covering approximately 15% of the exterior surface of the antenna. Of many paints considered, including epoxies, Hypalons, and acrylics, the acrylic paints (Krylon) had the best thermal radiation properties and the best adhesion to the envelope material.

Although the alodining provided the optimum surface treatment for temperature control, experience in fabrication with alodined foil indicated an embrittling of the foil resulting in a greater tendency toward a fracture of the foil and pinhole porosity when the material is sharply folded and flexed.

To prevent contaminating or otherwise altering the extensive surface treatment of the antenna materials, it became necessary for all fabricating personnel to wear lightweight nylon gloves while handling these materials.

After fabrication of the antenna assembly was complete, it was subjected to a series of evaluation tests, which included folding and packaging, deployment and inflation, dimensional tolerance, proof pressure and air holding. The folding test was obviously to assure that the complete antenna system could be folded into a container as described in a later section. The deployment and inflation tests were conducted in order to determine both the minimum time in which the antenna could be inflated to the design pressure of 10 inches of water and to check dimensional accuracy. Leakage tests or air holding tests were conducted to determine the air escapement from the inflated envelope, both to assure an adequate capacity for the inflation system and to evaluate the significance of possible disturbing torques in the transport vehicle as a result of the jet action of gas escaping through pinholes in the inflated antenna. It was determined that air leakage from the antennas was less than 10 cubic feet per hour. As a last test, the antenna was inflated to 15 inches of water for 10 minutes.

An environmental test program and operational inflation tests of the complete system in a space chamber are scheduled for the very near future.

PRESSURIZATION SYSTEM

In the initial design of the pressurization system, consideration was given to both compressed gas and subliming powders as methods for inflating the antenna. The compressed gas system was initially selected in preference to the subliming powder in order to obtain a more accurate control of the inflation and final pressure in the antenna. Preliminary

estimates originally indicated that an appreciable weight saving might be possible by the use of stored liquid carbon dioxide compared to storage of a high pressure gas. It is significant that while the resultant storage volume required for carbon dioxide is significantly less than that required for nitrogen, the relatively poor compression factor of carbon dioxide at 165° F. does not yield the resultant volume which would otherwise be expected using stored liquid. There is also a hazard, while using the carbon dioxide, of local momentary freeze-up in the expansion orifice during inflation.

For the above reasons it was, therefore, decided at this point that overall reliability would be enhanced by the use of stored nitrogen, and the program was continued on this basis.

A system was then designed which inflates and automatically deflates the 106 cubic foot antenna. Inflation is accomplished by releasing stored nitrogen gas through a series of explosive valves and orifices to the antenna. The inflation occurs in two stages. The initial inflation is through a small diameter orifice which slowly fills the antenna volume so as not to damage the envelope during deployment. After the initial inflation, or after the volume of the antenna is filled, a second explosive valve is actuated releasing gas through a large diameter orifice which quickly brings the antenna up to the design pressure of 10 inches of water absolute. Deflation of the antenna is automatically accomplished by means of bleed orifices in the antenna envelope which are sized to create a steady-state pressure in the antenna of 10 inches of water (inflow equals outflow) at the maximum inflation rate. The bleed orifices are located in nullifying positions on the antenna system so as to eliminate problems of disturbing torque on the transport vehicle.

CONTAINER

Based on folding tests conducted with early experimental antenna components, a container was designed which had inside dimensions of approximately 10" x 10" x 7". The container consisted of a bulkhead mounting plate to which the inflatable antenna was permanently attached and the covering canister, which is released just prior to inflation. The bulkhead plate contains the boom attachment to the launch vehicle, RF connectors for the coax cable to the antennas, an electrical connector for actuation signals and telemetry signals, the pressurization system, and antenna inflation fitting. Materials used in the fabrication of the container system were principally stainless steel and aluminum. Special provisions incorporated in the container system consist of a pressure-monitoring system to determine pressure inside the container, an air-tight seal between the canister and bulkhead mounting plate so that the canister can be evacuated, explosive latch pins for releasing the canister prior to antenna inflation, a teflon lining for the interior of the canister to protect the antenna during shock and vibration and to facilitate release of the antenna during deployment, and ejector springs to assure that the canister will be pushed from the bulkhead mounting plate at time of release.

Also incorporated in the design of the bulkhead mounting plates are two overlapping, folding arms, which are used to align the antenna, when inflated, with the bulkhead mounting plate. The arms are engaged to the support structure of the inflated antenna and the double-hinge arrangement provides for suitable folding of the antenna in the canister.

CONCLUSION

Although the program is not yet complete and the initial launch has not yet taken place, it has been demonstrated that an inflatable, active antenna for us in space environment is feasible and can be fabricated to extremely close dimensional tolerances. Range tests at sea level of the initial electrical model of the antenna system indicate that RF performance is within specification. Developments on this system indicate that the inflated structure concept is a uniquely advantageous means of putting large-volume structures into earth orbit having a minimum weight and for a relatively low cost.

October 9, 1963

ADVANCED RIGIDIZED MEMBRANE MATERIAL DEVELOPMENT

Presented At:

Conference on Aerospace Expandable Structures
Air Force Aero Propulsion Laboratory
Air Force Flight Dynamics Laboratory

Presented By:

P. H. Bratton, Vice President
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G. T. Schjeldahl Company
Northfield, Minnesota

DEVELOPMENT OF MATERIALS

FOR LARGE SPACE INFLATABLES

At the outset of this study, the requirements were set forth for an inflatable rigidizing passive communication satellite in joint effort by NASA and the G. T. Schjeldahl Company. The requirements discussed are separated into those concerned with 1) structural requirements, 2) availability, 3) processability of the materials, and 4) the basic radio frequency (RF) requirements.

Structural Requirements

The materials must be capable of being handled in a variety of mechanical and manual processes without damage from tearing, blocking, or stressing. They must also be undamaged in long periods of storage under such conditions as vacuums to one millimeter or less of mercury and temperatures of 90 to 100 F.

For deployment in space, the materials must withstand the shock of change from canister pressure of about one millimeter to space conditions of about 10^{-9} millimeter and must at the same time remain intact as the tightly packed unit expands from a small container of about ten cubic feet, to the full-sized unit -- a sphere several hundred feet in diameter which encloses a volume of several million cubic feet. In addition, they must not be damaged by local hot spots which may develop from direct exposure to solar energy and the albedo of the Earth. Since essentially no diffu-

sion of the radiation occurs around an object in space, the materials must also withstand the very low temperatures which are experienced by materials radiating to cold space.

After deployment, the structure must resist, without deformation, the forces arising from the pressure of solar radiation, impact of micrometeorite , and other minor forces arising from the albedo of the earth and electromagnetic effects. The requirements in this category are largely those of structural rigidity, but in initial phases of inflation, sufficient skin strength must be present to permit full inflation so that the projected shape is accurately obtained.

Availability

Materials available commercially in wide widths are highly desired if not required for programs where large size units are contemplated. It is particularly important that the materials be available in wide widths, preferably exceeding 48 inches so that a large number of square feet (125,000 for a 200 foot diameter sphere) can be reached with a relatively small number of pieces. The Echo II sphere, for example, 135 feet in diameter, has a surface area of approximately 57,000 square feet and is composed of 106 gores which are 48 inches wide by 212 feet long.

The importance of material made under commercial quality controlled conditions is highly important so that confidence can be established that the material used is not subject to defects and that it is of sufficient high quality for sample selection on a quality control basis. For this reason, space inflatables owe much to the existence of the highly developed packaging film industry in this country.

Processability

In general, materials which meet the requirements above are suitable for processing. However, sufficiently high tensile strength and modulus of elasticity is necessary so that processing will not deform the materials. In addition, it is necessary for the material to possess sufficient solvent resistance for the application of certain types of adhesives or to undergo heat sealing processes without developing areas which will adhere to each other.

To illustrate required handling, Figure 1 shows a part of lamination; Figure 2 shows a gore cutting operation, and Figure 3 shows the application of Alodine*, an inorganic conversion coating for the aluminum used in Echo II.

Since the purpose of the passive satellite is to furnish a Radio Frequency reflective surface, it was also necessary to meet basic RF

* American Chemical Company Registered Trademark, U. S. Patent Office

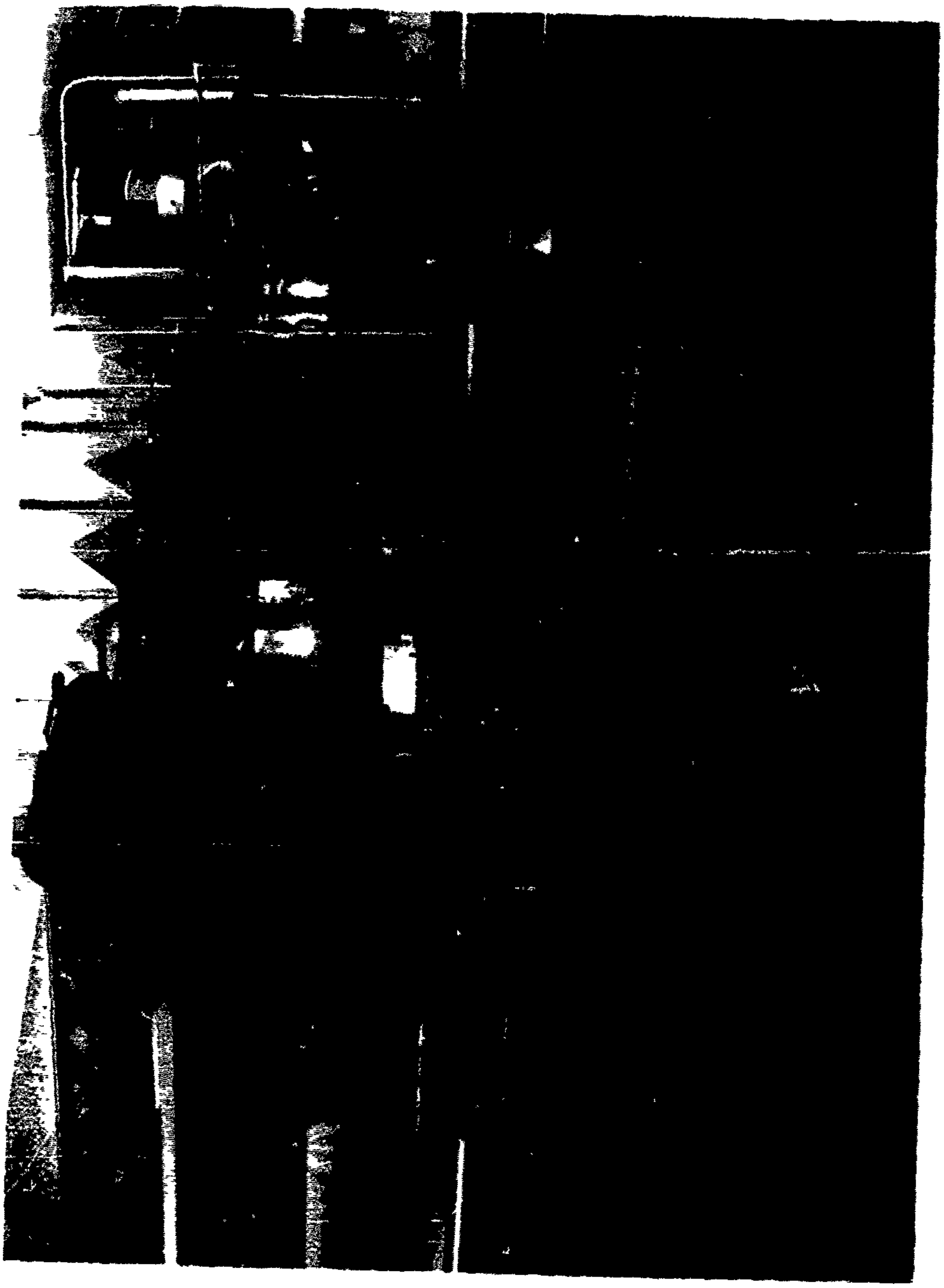


Figure 1

Figure 2

Figure 3

requirements. The concept studied began with the premises that the external surface should be continuously electrically conductive, and that the external surface could be a mesh-like aluminum structure with the mesh element spacings separated by a distance equal to some fraction of the wavelength of the RF energy reflected. The design wavelength was 3 cm and the spacing between elements was $\lambda/8$ or 0.375 cm (about 0.17-inch).

Materials Selection

At the outset of this investigation the behavior of Echo I (fabricated from 0.5-mil metalized Mylar) was known to have resulted in a somewhat deformed sphere after the first several weeks in orbit. This resulted from 1) loss of inflatant material due to puncture by particles in space, 2) the fact that the orbiting system had thermal balance characteristics which allowed the temperature to range up to about 150 C and 3) the fact that the Mylar used had associated with it plastic memory for its former rolled condition and folds. Information was also available from early testing of Echo II which indicated that the projected material would be more rigid than necessary and that it would be sufficiently temperature resistant because a thermal balance coating was to be applied to maintain the temperature below 100 C. At this point it was proposed that a material be developed which would be superior to Echo I and Echo II material. This was to be accomplished by reducing the cross section area offered to solar pressure which constituted the major force tending 1) to deform the sphere and 2) to cause the sphere to change its orbit.

Initial studies of the literature and work with the packaging industry indicated that the two most promising materials for the basic plastic structure were Mylar* and UDEL** biaxially oriented polypropylene which was just then becoming commercially available. The

* E.I. du Pont Company Trademark, Registered U. S. Patent Office
** Union Carbide Plastics Company Trademark, Registered U. S. Patent Office

properties of these two materials are shown in Table I and indicate that polypropylene possesses a somewhat better strength-to-weight ratio but its tear resistance is only approximately 30 to 40 percent that of Mylar. Initial interest in biaxially oriented polypropylene was intensified as the feasibility of widths to 60 inches and more was definitized. Intensive work was done on UDEL biaxially-oriented polypropylene by the Union Carbide Corporation.

TABLE I

PROPERTIES OF MYLAR AND POLYPROPYLENE

PROPERTY	MYLAR	POLYPROPYLENE
Tensile Strength (psi)	20,000	20,000
Modulus of Elasticity (psi)	5×10^5	2.5×10^5
Density (gm/cm)	1.39	0.91
Tear Strength (gm/mil)	15	5
Availability (inches)	72	60
Minimum Thickness (mil)	0.25	0.35 - 0.5
Strength to Weight Ratio (inch)	4×10^5	6×10^5

The initial concept was based largely on the desire for a very high safety factor in terms of the ratio of the pressure required to destroy the device to that required for inflation to produce rigidity. For this reason, emphasis was placed on films of approximately 0.35- to 0.5-mil thick. This led to the concept of developing the "I" beam type structure shown in Figure 4. This sandwich type structure was projected to make maximum utilization of the thickness of plastic believed at that time to be necessary and to rely on the strain hardening of aluminum to obtain rigidity.

The concept of strain hardening of aluminum was demonstrated in the early tests of the Echo II material and at the outset the modulus of aluminum, i.e. about 10×10^6, was regarded as high enough to overcome the effects of Mylar with a modulus of 5×10^5. Experimental work, however, indicated that the modulus of aluminum in very thin foil (180 microinches) is apparently a third to a quarter of that for the bulk material. Sufficient rigidity can be obtained, however, for configurations in which complete exposure of the plastic does not occur.

Structural Analysis

With these facts in mind, structural analysis studies were conducted to define in general terms the effect of etching various geometrical

"I" BEAM EFFECT ILLUSTRATED

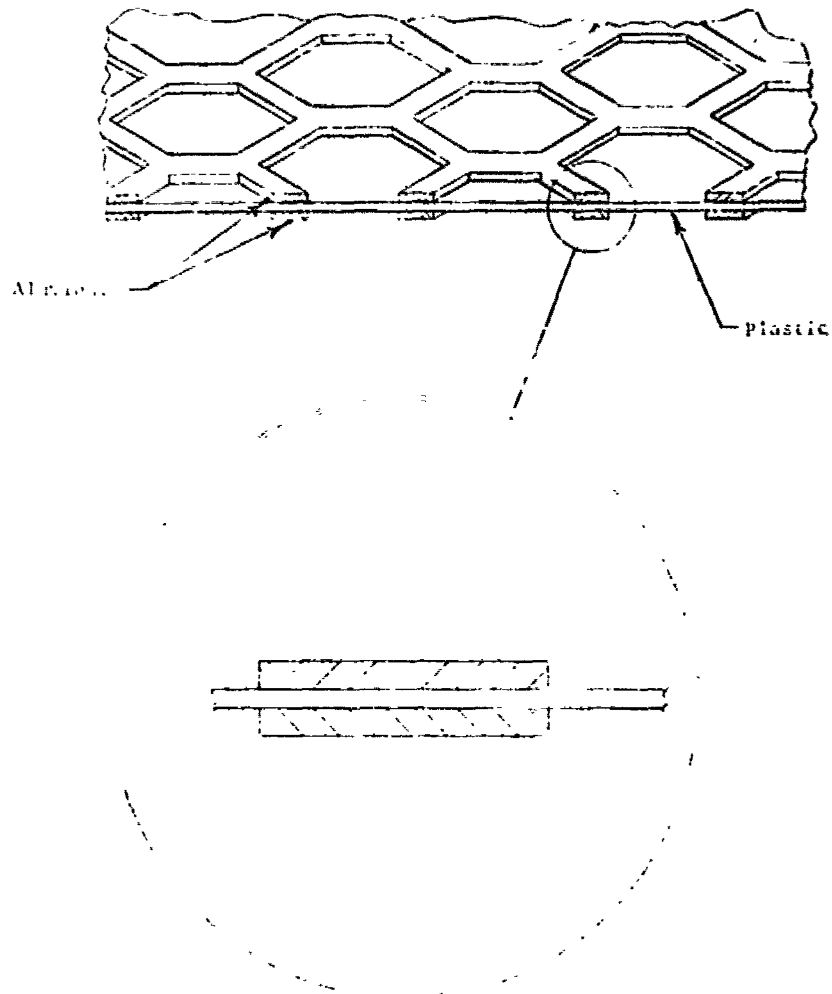


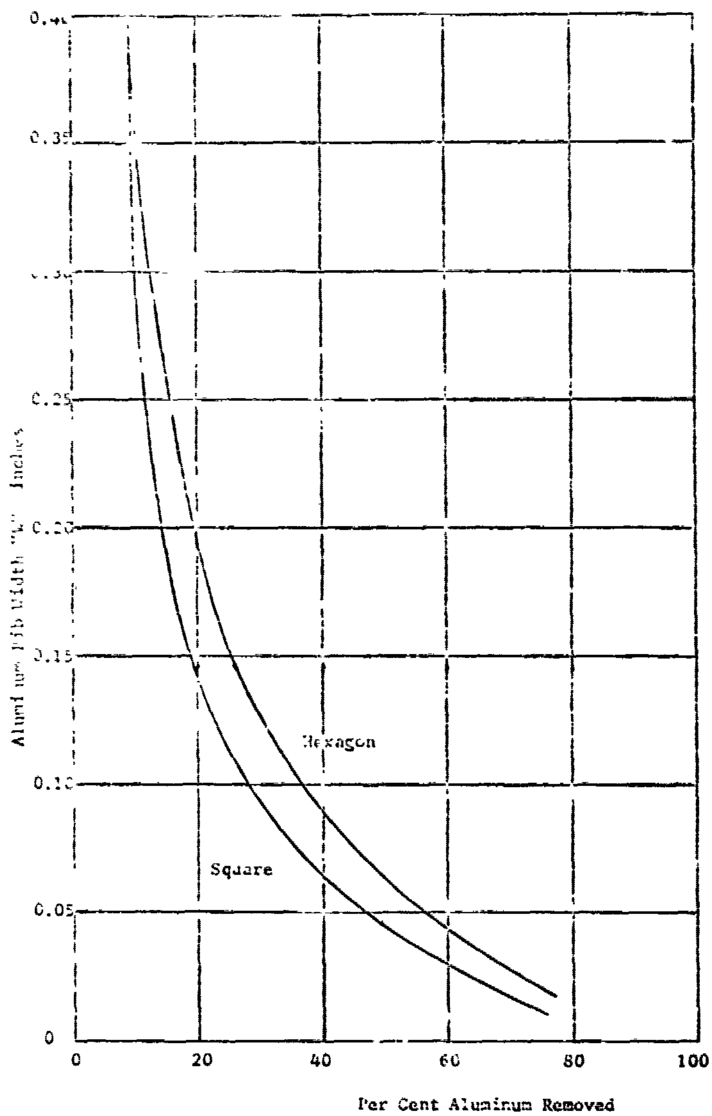
FIGURE [illegible]

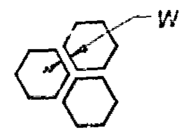
patterns, various ratios of plastic thickness to aluminum thickness, and various plastic materials on the final properties of the material. In this work, the analytical expressions developed by the techniques shown for the hexagonal pattern, Figure 5, showed that for a given fraction of aluminum removed, the hexagonal pattern would result in the widest rib of remaining material, indicating that it would also yield the greatest strength to weight ratio. Therefore, the hexagon was selected for intensive study because its three directional orientation represented a closer approach to anisotropic properties than is achieved in a square pattern. Early laboratory experiments indicated that, while this might not be exactly true, no serious sacrifice was made in the selection.

Tests were then run with laminates of Mylar and polypropylene with various thicknesses of aluminum foil and many were made into etched configurations through the use of silk screen and photographic resist processes. Etching was done with ferric chloride in the early phases of the investigation.

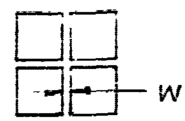
Conventional stiffness measuring methods were examined for characterization of these materials. However, standard compression tests were not applicable and the usual Taber stiffness did not yield data which were compatible with the standard theories of beam rigidity.

Figure 5





$$W = \frac{0.15}{\frac{1}{\sqrt{M}} - 1}$$



$$W = \frac{0.15}{\sqrt{2}\left(\frac{1}{\sqrt{M}} - 1\right)}$$

m = Fraction of Aluminum Removed

This situation had been anticipated at the outset of the program and a number of five-foot diameter hemispheres were fabricated for rigidity studies of spherical sections. These were carried out as shown in Figure 6, where the hemisphere was tested in both up or down position. In some cases, the pole cap was in the center of the hemisphere and in others two pole cap halves were at the edges of the hemisphere. These experiments indicated that the rigidity of the etched laminates to be much less than that of the parent materials but still sufficiently great for use in the space weightless conditions. However, the need for information which could be translated in terms of known test conditions was felt to be necessary. For this reason a number of experimental methods were devised in an attempt to define the properties of the laminates. Such methods as deflection of small tubes and conical loading of formed and sealed cylinders showed promise. The latter method, shown in Figure 7, gave data consistent with theory and indicated that the strength of the etched laminate, while significantly below that of the unetched laminate, was sufficient for their use under space conditions. As a first step, the technique was used to correlate theoretical calculations and experimental data for non-etched material. Then the etched material was compared to non-etched material to obtain an empirical factor which was applied to the equations developed for the large radius-thin shell structure under consideration.

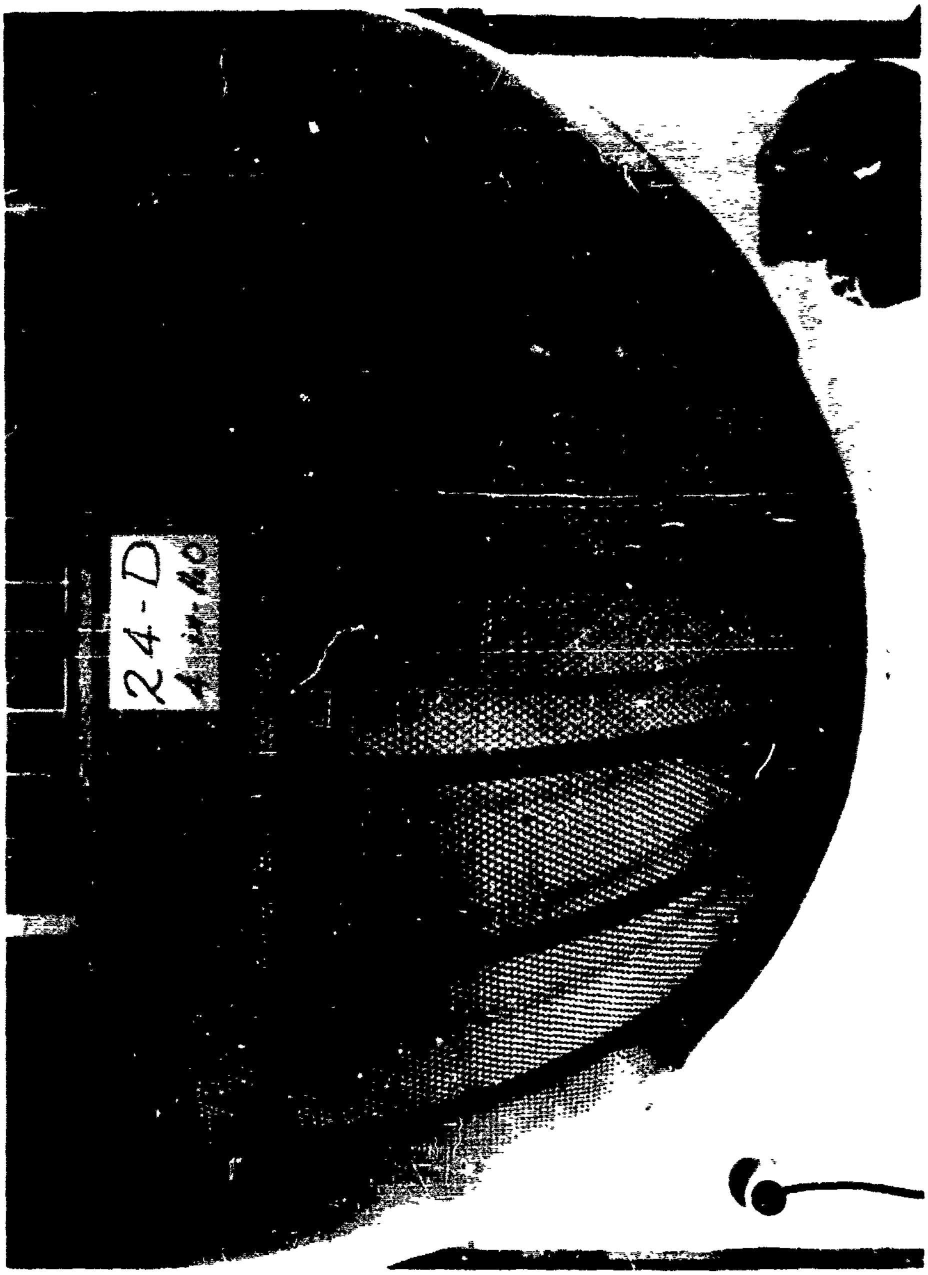


Figure 6

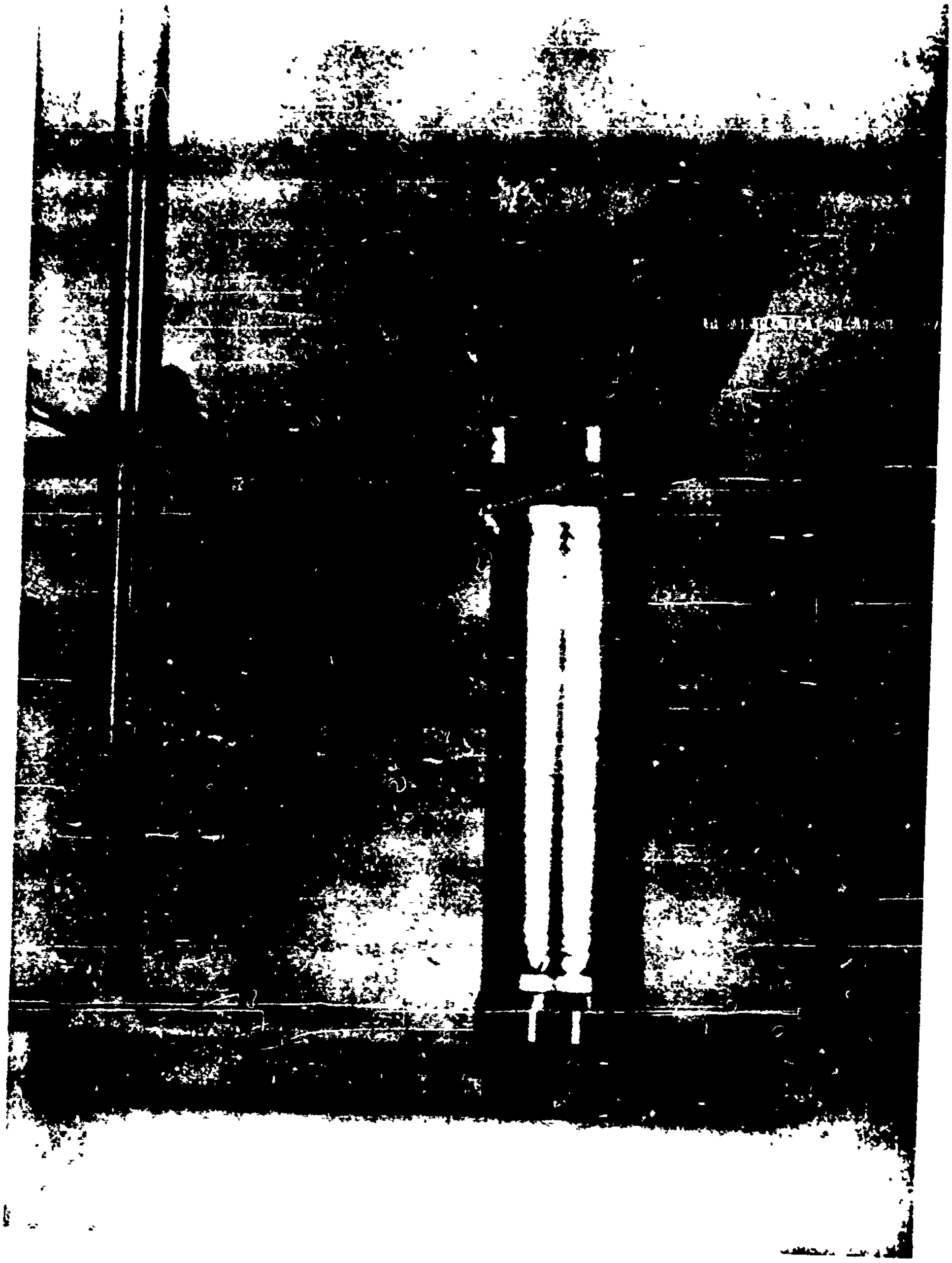


Figure 7

The equation developed for local buckling of the thin shell is as follows[1]:

$$P_{cr} = \frac{E \mu t^3}{30\ R\ (\pi r^2)}$$

where: P_{cr} = Critical buckling pressure for local buckling,

E = Modulus of elasticity,

μ = Empirical dimensionless parameter,

t = Material thickness,

R = Sphere radius

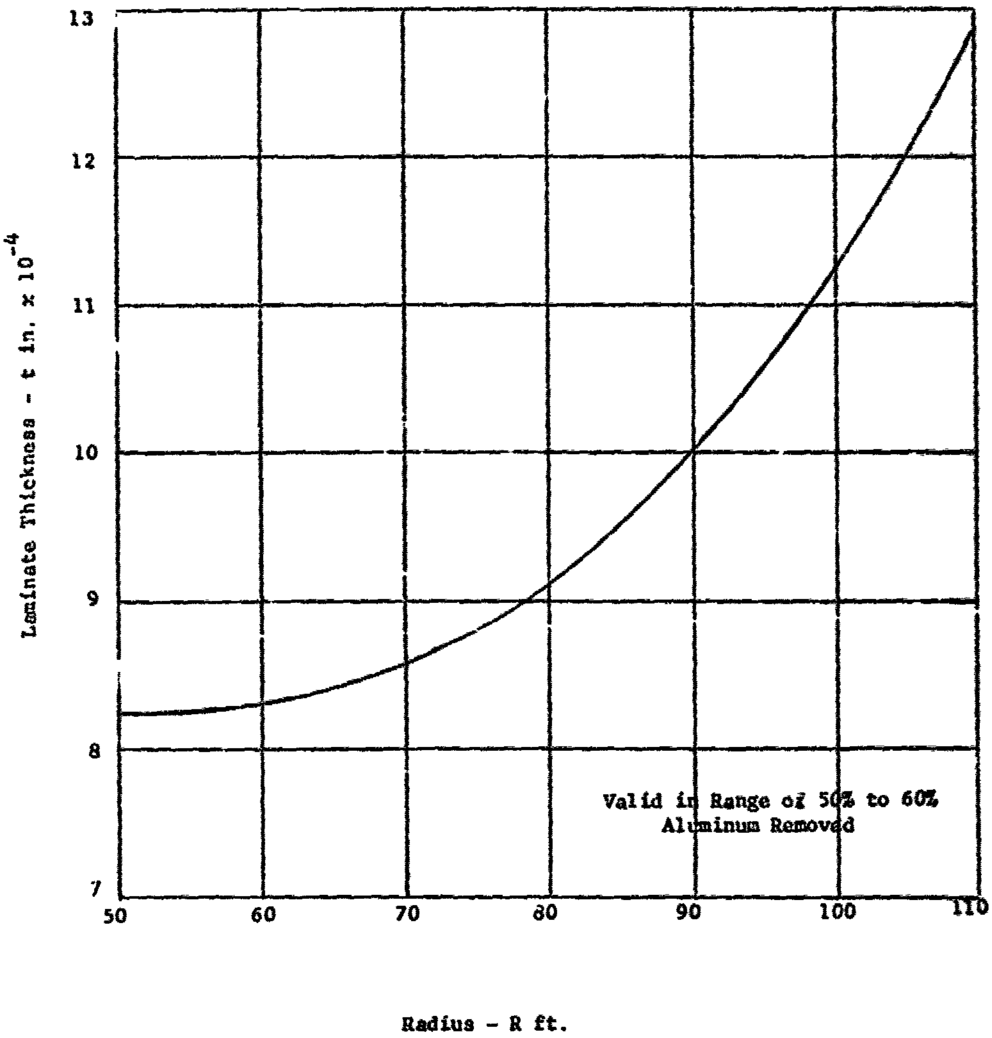
r = Radius of deformation in local buckling.[2]

The results of this analysis were compared to other analytical approaches which included the calculation of the critical stress for a small spherical cap and the calculation of critical stress of curved beams where each of the mesh ribs were considered a curved beam. In general, the various approaches were in good agreement and therefore the analysis was applied with confidence. The equation is described graphically in Figure 8.

[1] Technische Dynamik, C.B. Biezano and R. Grammel, 1939.

[2] "Axially Symmetric Buckling of Shallow Spherical Shells Under External Pressure", [illegible], Dec. [illegible].

Laminate Thickness vs Sphere Radius
Aluminum - Polypropylene
Milled Laminate



Fabrication Processes

In the experimental fabrication work, methods were devised for laminating aluminum to biaxially oriented polypropylene using the processes similar to those already established for aluminum and Mylar, Both systems used the GT*-301 Schjelbond* system which was space proved in Echo I. A routine test for both the laminates and seals consisted of simultaneously dipping a sample and rotating it around a 0.125-inch mandrel, alternately from liquid nitrogen to boiling water. This test method demonstrated the utilities of the materials. It is particularly interesting that oriented polypropylene retains its flexibility at low temperature despite the rather high brittle point encountered with unoriented films of the same material. It should be noted that the development of satisfactory bonds to polypropylene with the adhesive system required a corona treatment prior to application of adhesives.

The interim production of materials for study on larger scale was translated from the laboratory techniques of silk screen and photo-resist to rotogravure printing. Experimental work carried out by Marathon Printing Company demonstrated both the feasibility of the method and the suitability of an ink devised by them for the etching process envisioned. This ink had sufficient resistance to strong

* G. T. Schjeldahl Company Registered Trademark, U.S. Patent Office

sodium hydroxide solutions so that it could serve to preserve the desired pattern in the etching process. The rotogravure printing was ultimately carried out to provide continuous rolls of material printed on both sides in register with patterns containing 0.15-inch hexagons with 0.03-inch wide lines. Material from this printing and from 20-inch webs was etched in continuous processes, (see Figure 9).

Model Sphere Tests

As detailed information was accumulated on strength properties of the materials, a large number of small spheres were made to demonstrate the suitability of the materials for both handling and deployment. It is interesting that the initial units made failed in numerous ways which indicated minor processing changes were needed to insure higher reliability in the final 20-foot model spheres. The test carried out on 30-inch and 12.5-foot spheres included inflation at room conditions to burst, exposure to high temperature on one side and low temperature on the other in environmental chambers under vacuum at Wright Field, and deployment of 30-inch spheres from a small model canister in a vacuum chamber. Figure 10 shows a 30-inch sphere in the vacuum chamber.

DIAGRAMMATIC REPRESENTATION OF ETCHING AND ... REMOVAL PROCESS IN ALODINE COATER

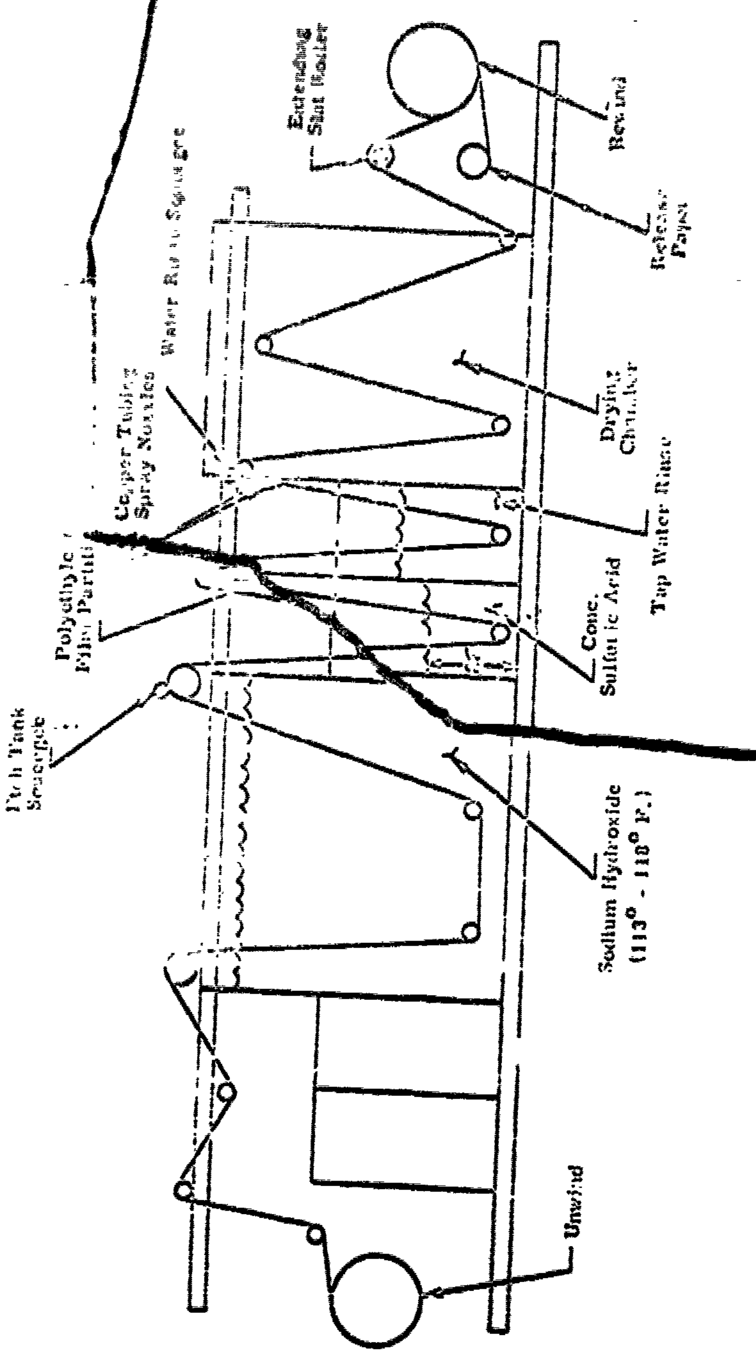


Figure 9

In the final phases of the initial work several 20-foot diameter spheres were made, packed, and folded into model canisters for vacuum deployment and inflation tests at Langley Research Center. Under conditions of about 30 microns pressure and 60 to 70 F the model spheres deployed and inflated rapidly to full diameter (one to two seconds). The inflatant materials used in these experiments were acetamide and ammonium carbonate. The latter material was one which was selected under the current program as offering minimum effective molecular weight for use in space inflatables. A 20-foot sphere in the [illegible] after ejection from the canister is shown in several stages in Figures 11 through 14.

The full 20 foot spheres indicated that certain minor problems, associated largely with thermal balance and the possibility of blocking unless complete removal of adhesive from the surface of the sphere was insured still existed. The latter point developed since one sphere showed tears because a small amount of adhesive from a seam adhered to another portion of the sphere and caused tearing during deployment. Simulated beacons present during these tests did not appear to lead to any damage on deployment and inflation.

Behavior of Plastic Materials Under Space Conditions

Throughout the present program the behavior of the plastic materials under space conditions had been of great concern, since the original

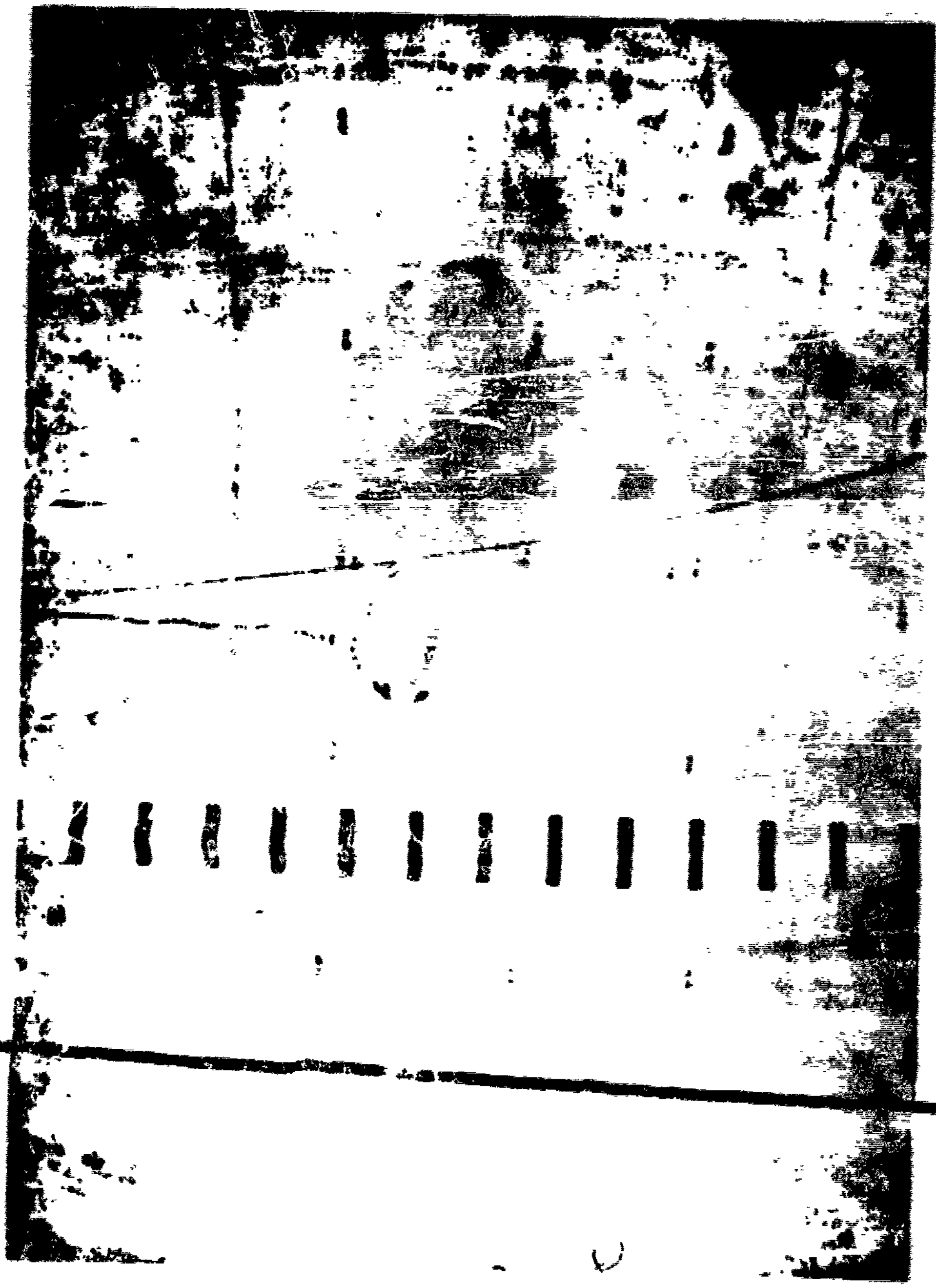


Figure 11

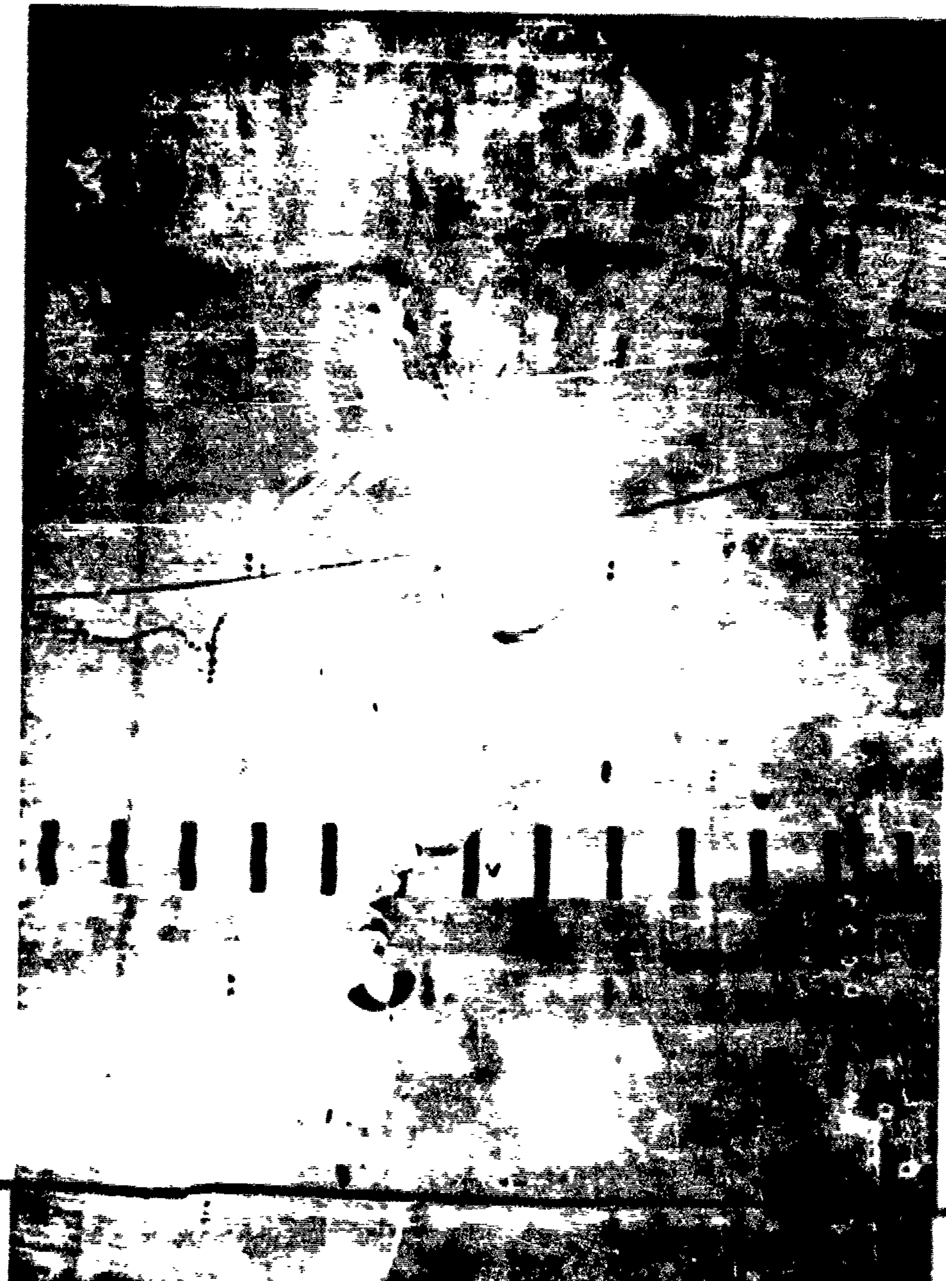


Figure 12

Figure 13

Figure 14

effort was directed toward the use of a material which would ultimately evaporate or disappear under space conditions. As the program proceeded and initial tests were made it became apparent that the disappearance of plastic under space conditions was not an easy situation to achieve. For this reason, it was thought that a suitable solution to the problem could be realized by employing a plastic film which would remain transparent to solar radiation and act as a window rather than as a hole. This circumstance would also lead to a minimum effect of solar radiation pressure on the sphere since pressure is exerted only when the radiation is reflected or absorbed.

The efforts designed to measure the tendency of plastics to degrade under space conditions were carried out at St. Olaf College in Northfield. Additional experimental work was done under subcontract by Radiation Applications Inc., Long Island City, New York. This work indicated that polypropylene, while known for its ease of degration in ultra-violet light in the presence of oxygen, was extremely resistant to weight loss and even to change in physical properties when exposed to vacuums of 10^{-4} to 10^{-5} millimeters of mercury in the presence of radiation from a General Electric UV-6 mercury vapor lamp. Exposures for periods up to 68 days indicated that 0.5-mil biaxially oriented polypropylene gave slight increases in yellowness, and became somewhat

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0.35 Mil [illegible]

Cont[illegible] Sample Versus Sample [illegible] U. V. and Vacuum for 62 days

Cary 14-[illegible]

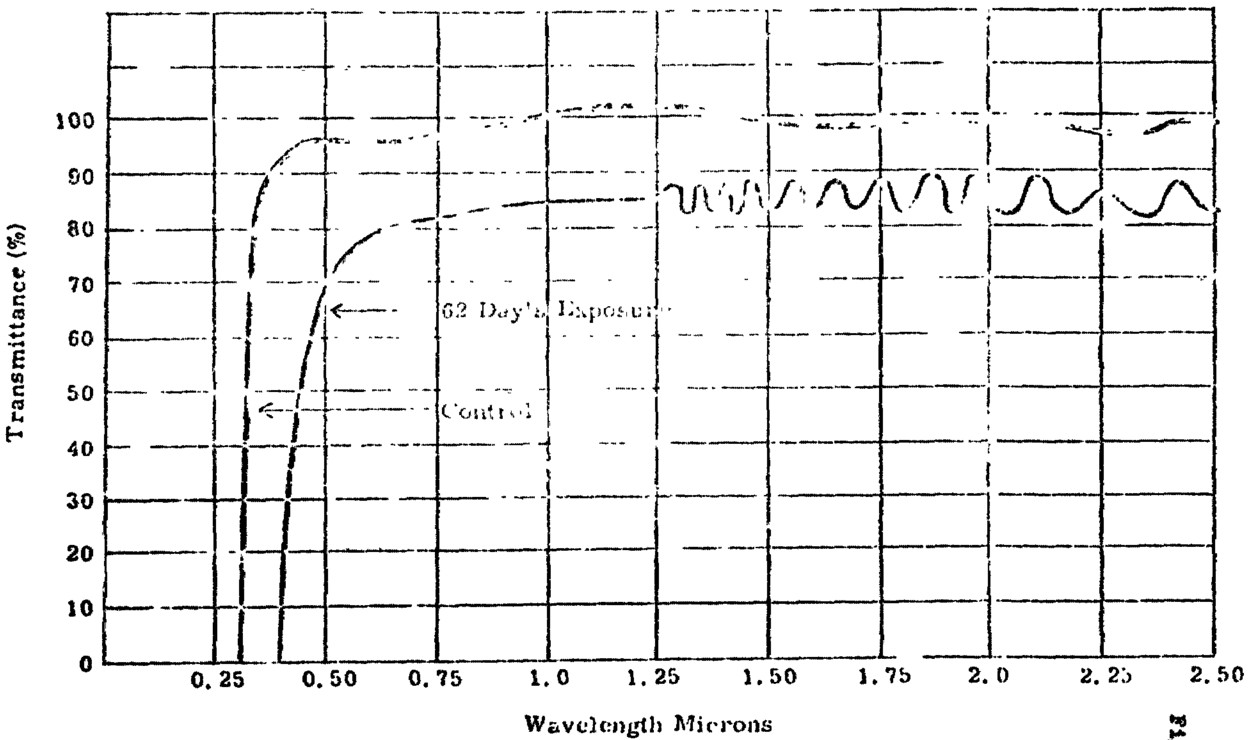

Figure 15

[illegible]
[illegible] V, and Vacuum for 62 days
[illegible]

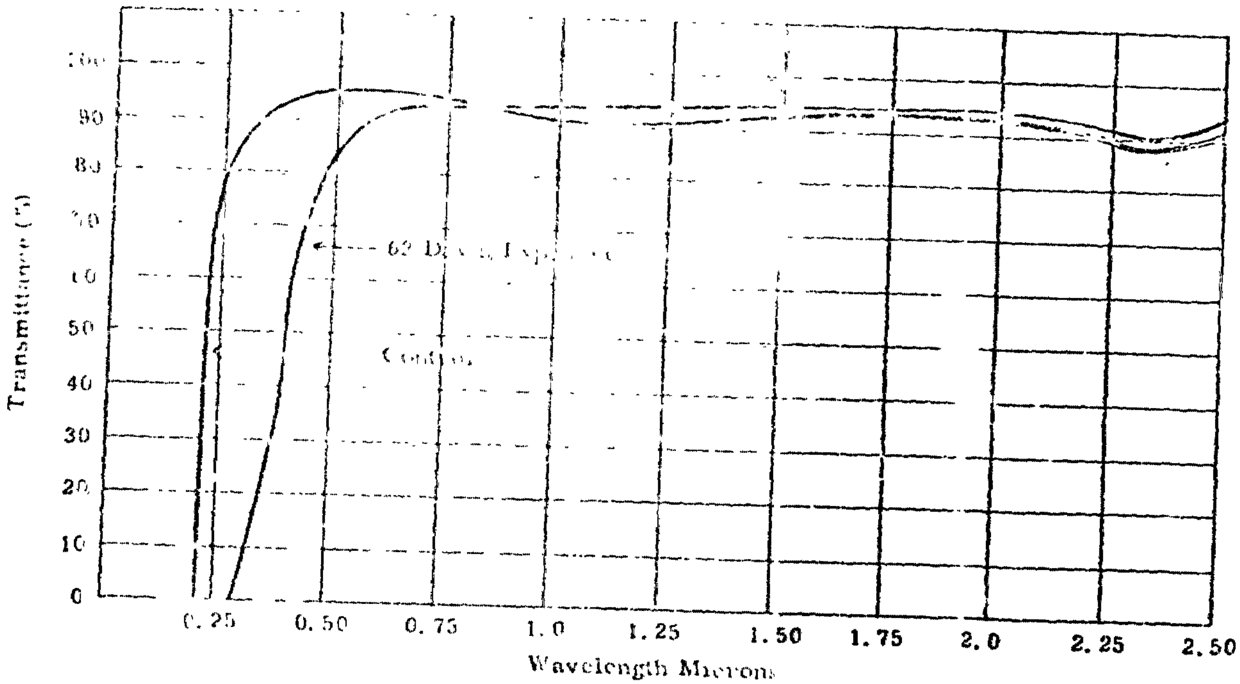

Figure 16

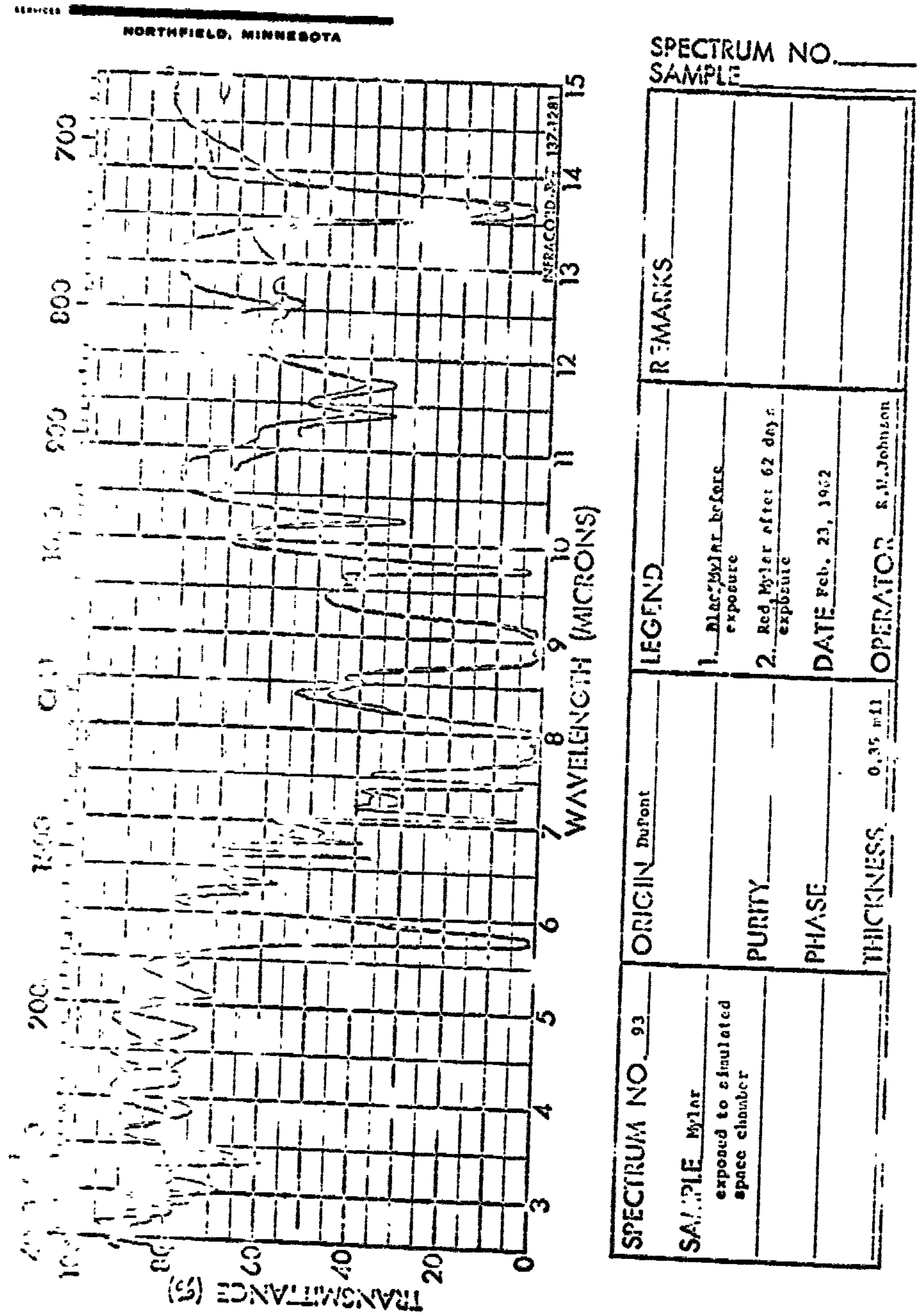
NORTHFIELD, MINNESOTA
WAVELENGTH (MICRONS)
TRANSMITTANCE (%)
SPECTRUM NO. 93
SAMPLE Mylar
exposed to simulated space chamber
ORIGIN DuPont
PURITY
PHASE
THICKNESS 0.35 mil
LEGEND
REMARKS
DATE Feb. 23, 1962
SPECTRUM NO.
SAMPLE

Figure 17

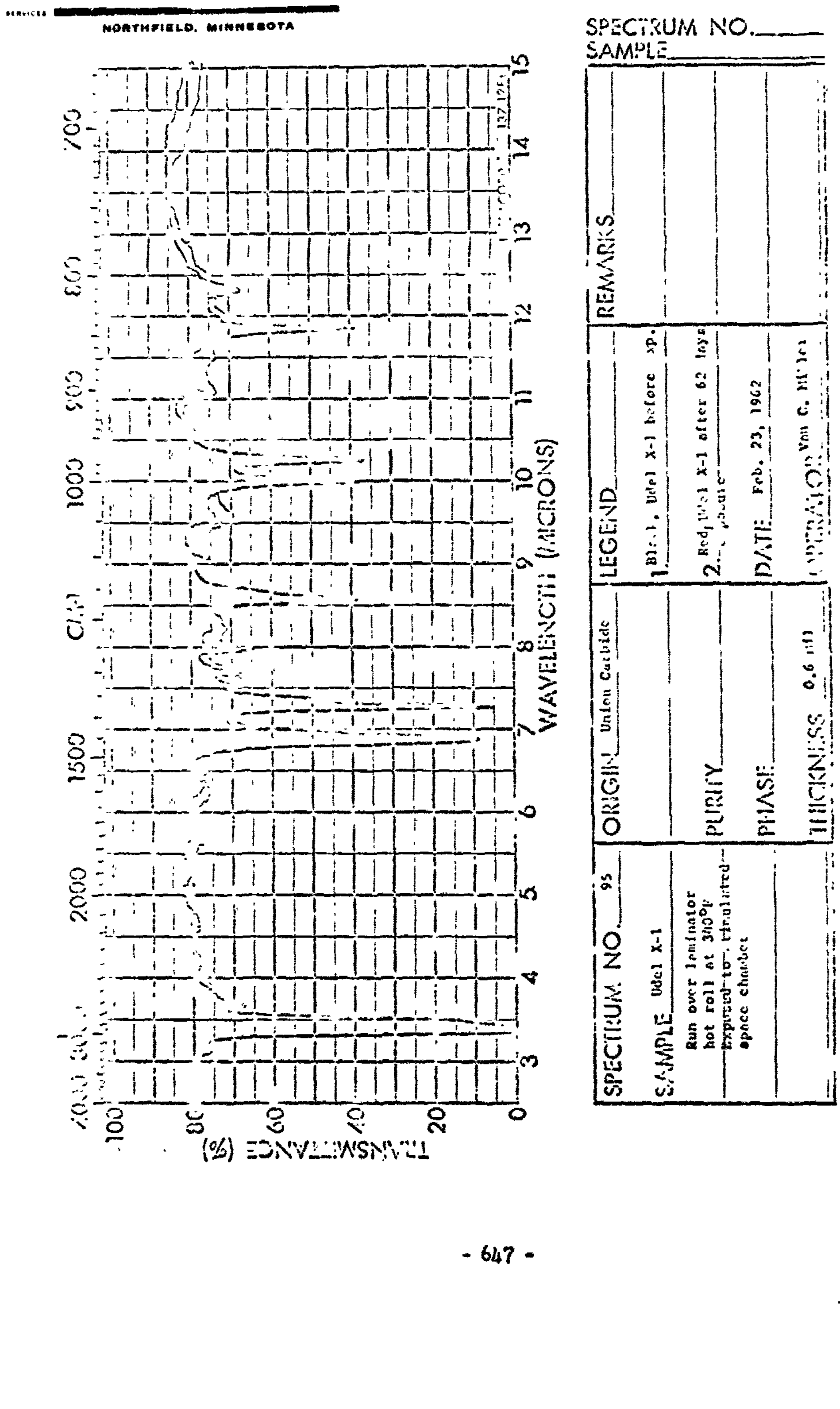
NORTHFIELD, MINNESOTA
SPECTRUM NO.
SAMPLE
WAVELENGTH (MICRONS)
TRANSMITTANCE (%)
SPECTRUM NO. 95
SAMPLE Udel X-1
Run over laminator hot roll at 340°F
ORIGIN Union Carbide
PURITY
PHASE
THICKNESS 0.6 mil
LEGEND
DATE Feb. 23, 1962
REMARKS

Figure 18

Figure 19

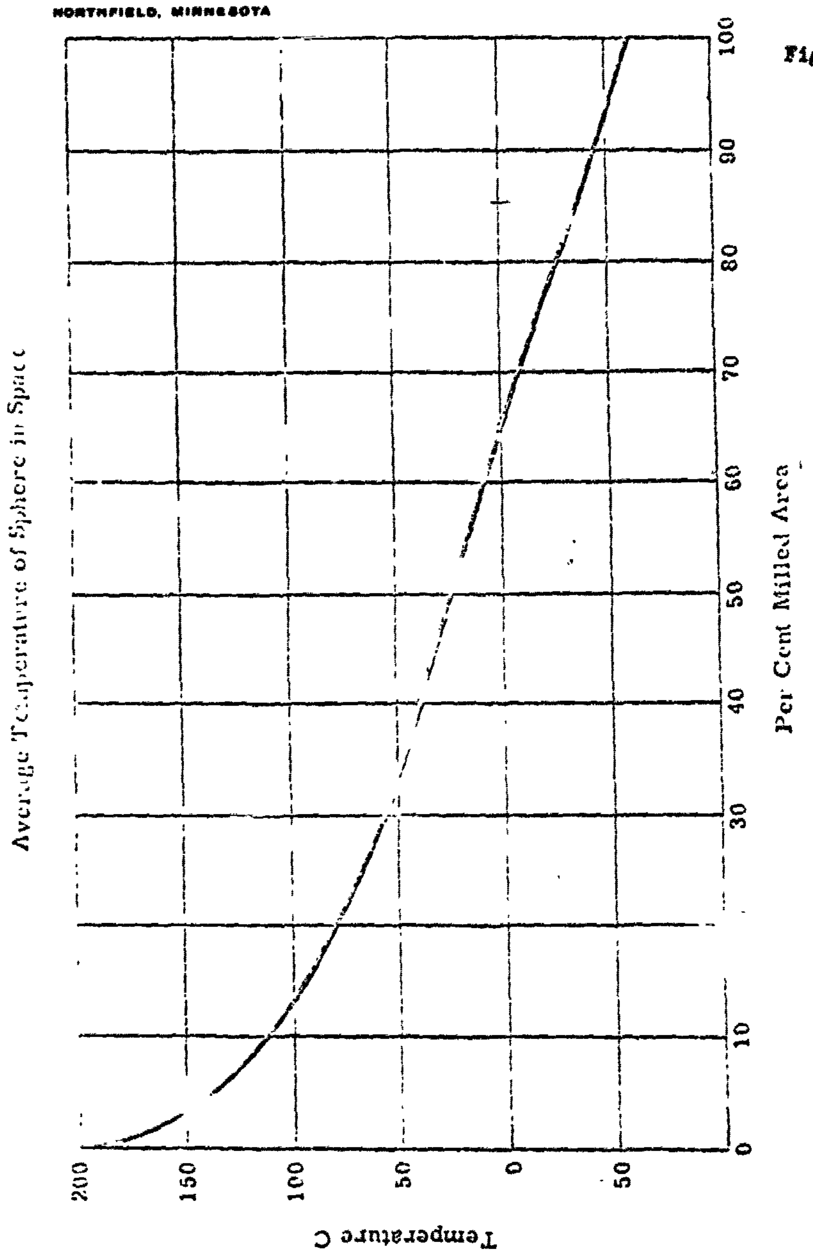
NORTHFIELD, MINNESOTA
Average Temperature of Sphere in Space
Per Cent Milled Area
0
10
30
40
50
60
70
80
90
100
Temperature C
200
150
100
50
0
50

TABLE II

TEMPERATURE SUMMARY

(in line between earth and sun)

CASE	I	II	III	IV	V
Al Removed (%)	70	58	70	58	70 coated*
Polypropylene	Unexposed**	Unexposed	Exposed***	Exposed	Exposed
Temp.Max. (C)	73	97	139	149	71
Temp.Min. (C)	-18	7	30	39	-3
Temp.Avg. (C)	-8	12	42	51	-2

* White enamel, TiO_2 pigment

** Polypropylene not exposed to UV or vacuum

*** Polypropylene exposed to both UV and vacuum

CONCLUSIONS

The aim to design an advanced material for inflatable rigidizing satellites has been achieved and suitable design parameters defined. Spheres of this material will operate very satisfactorily as passive communications satellites.

Polypropylene was the material chosen for fabrication of a space-inflatable rigidizing satellite because of its low density, good tensile properties, and low color buildup in a space environment. Spectral analysis of polypropylene over most of the range covered by solar radiation in outer space has shown very low absorption of the sun's energy. Therefore, the need for a "disappearing" plastic film for this application is much less important than originally thought in view of the high spectral transmission of polypropylene. Solar radiation will exert very little pressure on a film transparent to 90+ percent of the sun's energy.

The best overall adhesive system for laminating aluminum foil to polypropylene film was Schjelbond GT-301, a proprietary G. T. Schjeldahl Company thermosetting resin adhesive. In order to obtain the bond to polypropylene, the film should be exposed to a low-level corona treatment.

Handling during fabrication, packing, deployment, and inflation has no adverse effect on the structural qualities of these advanced

materials. The materials are not subject to blocking during prolonged compressed storage at elevated temperatures.

The materials developed under this program are better able to withstand rapid inflation in space conditions than are present materials used for large-diameter inflatable spheres.

The rotogravure printing method and the associated sodium hydroxide etching was developed to the point where registration of the patterns from one side to the other in printing and etching was achieved on a continuous basis with 54-inch wide material. In general, the printed resist method appears to yield a very satisfactory etched material.

A theoretical approach and design equations have been developed for the study of temperature conditions for a sphere in space. The theory shows that the sphere can be kept cool by controlling the percent of area milled or by applying a thermal coating or by a combination of these two factors.

A sphere 200 feet in diameter with a total weight less than 900 pounds is feasible with a safety factor of 3.0 for rigidity. The milled laminate weight is comparable to Echo I material at sphere diameters of 135 feet or less and it is less than 60 percent that of Echo A-12 laminate for a 135-foot diameter sphere.

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EXPANDABLE STRUCTURES
- RECOMMENDATION FOR THE FUTURE -

- R. B. Liddell
- S. L. Lieberman

Spacecraft Organization
Lockheed California Co.

24 October 1963

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- RECOMMENDATIONS FOR THE FUTURE -

By - R. B. Liddell
S. L. Lieberman

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INTRODUCTION

The growth of expandable structures for aerospace application has been quite rapid during the past four years. Previously, this activity centered around such conventional devices as balloons and parachutes. Since many of those present played significant roles in the subsequent acceleration in the state of the art, there is no need to dwell further on this period. However, the causes and consequences of this rapid growth deserve some elucidation.

The obvious requirements for minimum packaged payload launch weight and volume lent impetus to the early development, design, and fabrication of expandable structures for such applications as extremely large diameter (e.g. 100-135 foot) passive satellites, and mechanically foldable or retractable booms and legs for unmanned orbital (e.g. NIMBUS) satellites and space traversing (e.g. RANGER) vehicles. The collapsible booms offered a particular versatility to the orientation and location of solar "paddles" and magnetometers, in addition to the more prosaic antenna applications.

It soon became evident that the pressurization of spheres and mechanical extending of devices from the main structure, were merely simple precursors to the larger technological area; expandable structures for the aerospace environment. As the aerospace needs became diversified, so did the methods of achieving them via expandable structures. The techniques presently range from the rather self-evident concepts noted above, to increasingly more sophisticated ones as typified by the use of irradiated plastics with a memory, or the deployment and rigidization of vapor catalyzed systems; the simplicity of the former and the complexity of the latter, not withstanding. As a result of the increasingly broad application framework for these highly packageable structures, various R & D programs have been initiated by applicable government agencies and private organizations to solve specific aerospace problems via expandable structures.

In order to avoid duplication of effort by the interested organizations, and to determine the most important and potentially most immediately fruitful areas of R & D, the Spacecraft Organization of the Lockheed California Company was requested to carry out a "Study of Expandable Structures" for aerospace applications for the Air Force.

PROGRAM OBJECTIVES AND METHODS

It was the objective of this program to compile the available data on the subject and carry out an assessment of the state of the art. The general program parameters were:

1. Emphasis on pertinent materials and processes for candidate concepts.
2. Comparison of first generation expandable structures.
3. Mission limitations:
   a. Earth reentry from low altitude (i.e. 300-500 n.m.) circular orbital velocities of approximately 25,000 feet/second.
   b. Non-reentry space operations at near-earth orbital altitudes.
4. No effort on new materials, processes, or structural concepts.
5. Exclusion of classified data.

The general method of conducting this program was straightforward. However, it should be noted that the compilation phase involved the personal visitation to 74 organizations and the collection of over 1200 documents. In assessing the information, it was necessary to arbitrarily classify the quality of the data based upon a preference of source. The selection was made, in order of preference, from:

1. Space data - Derived from recovered space objects, space telemetry, or from space phenomena observed from earth.
2. Data obtained in a simulated space environment - Preferably derived from simulation of more than one environmental space factor at a time.
3. Earth data extrapolated to a space environment -
4. Earth data -
5. Derived theoretical data -

Literature based on genuine space data was, as might be expected, quite scarce, whereas literature based on derived theoretical data was plentiful. The quality of the information was often uneven and sometimes even dubious. For example, authorities on micrometeoroid hazards differed in their estimates by several orders of magnitude. Environmental information was generally derived from secondary sources such as recognized authorities rather than directly from experimental data. Data on ascent, orbital, and reentry conditions were based upon assumed vehicle configurations and trajectories.

Upon completion of the data assessment, the information considered useful to a designer of expandable structures was incorporated into a handbook. This handbook presents data on the aerospace environment to be encountered during ascent, orbiting, and reentry. It treats first generation concepts in some detail. Applicable structural analysis techniques are included for the convenience of accessibility to the designer. Although the basic analysis is essentially no different than for other, more conventional structures, significant modification was sometimes required to develop a complete operational or "working" analysis, as in the case of AIRMAT*. The properties of materials

* Goodyear Aircraft Co.

presently considered of interest in designing expandable structures are presented, as well as a review of pertinent fabrication and processing techniques. This includes methods to carry out the deployment and rigidization processes.

A final report was also prepared containing a discussion of the program, its findings, various conclusions and recommendations, and, finally, a bibliography of over 1200 documents.

CONCLUSIONS AND RECOMMENDATIONS

As a consequence of reviewing these documents and the information obtained during the organizational surveys, a number of conclusions and recommendations concerning aerospace expandable structures were defined; some being self-evident, others not as apparent. For convenience of analysis herein, these findings can be categorized as follows:

1. Applications
2. Structural Concepts
3. Materials - Primary and Ancillary
4. Processing

APPLICATIONS - The possible utilization of expandable structure concepts for almost any conceivable aerospace application may someday be feasible. However, from the standpoint of practical design, a realistic appraisal must be made of present needs versus present capabilities, available design data, and the possibility of preferable non-expandable concepts. Immediately useful applicability involves the missions noted in Table 1.

Table 1

AEROSPACE EXPANDABLE STRUCTURES
FOR MAJOR MISSIONS
(PRESENT DESIGN REQUIREMENTS)

	MISSION	TYPICAL APPLICATIONS
Atmospheric Reentry	Booster Recovery	SATURN C-5
	Vehicle Recovery	Space Station Logistics Vehicle
	Personnel Recovery	Emergency Return From Orbital Satellite Or Space Vehicle
Non-Reentering Space Operations	Energy Reflectors/Radiators/ Collectors	Passive Satellites, Solar Collectors, Antennas
	Structures	Extension Devices and Space Shelters

However, in order of present apparent mission interest, they are:

- o Energy Reflectors/Radiators/Collectors
- o Booster Recovery
- o Structures
- o Vehicle Recovery
- o Personnel Recovery

STRUCTURAL CONCEPTS - In general, the program disclosed an abundance of significant concepts to carry out the above missions. The pursuance of the various concepts not only varied with particular organizations interests and capabilities, but was also susceptible to overall aerospace objectives of the Air Force and NASA. This is clearly seen in Tables 2 to 4, which indicate the usefulness of specific expandable structure devices with respect to the major missions of immediate interest.

Table 2

AEROSPACE EXPANDABLE STRUCTURES
FOR REENTRY RECOVERY
- BOOSTERS, VEHICLES, AND PERSONNEL -

Parachute	Rotorchute
Drag Brake: Petal	Paraglider (Parawing, Rogallo Wing, Flex Wing)
Drag Brake: Variable Area (Umbrella)	Unfolding Wings
Drag Brake: Inflatable Struts	Inflatable Wings
Ambient Air Bag (BALLUTE)	Honeycomb Wings
Ambient Air Bag (ROOST)	Foamed Wings
Ambient Air Bag (Landing)	Foam Encapsulation (MOOSE)
Hot Air Bag (PARAVULCOON)	

Table 3

AEROSPACE EXPANDABLE STRUCTURES
FOR NON-REENTERING SPACE OPERATIONS
- ENERGY REFLECTORS/RADIATORS/COLLECTORS -

Inflated Balloon: ECHO I	Inflatable Spiral Boom
Yielded Metallic/Plastic Balloon: ECHO A-12	Yielded Metallic Foil Boom
Inflated Rod	Multiple Hinge Linear Array (Accordian Fold)
Inflated Tubes/Membrane: NABLA	Metal Mesh (Spin-up Deployment)
Membrane Stiffening via Toroid Rigidization (Pressurized, Chemical, or Physical)	Inflatable Solar Shield: AIRMAT
Inflated Double Paraboloid	Erectable Solar Shield: Parasol
Inflated Lenticular	Solar Concentrator: Petals
	Solar Concentrator: Foam
	Stacking Ring Parabola

Table 4

AEROSPACE EXPANDABLE STRUCTURES
FOR NON-REENTERING SPACE OPERATIONS
- STRUCTURES (EXTENSION DEVICES & SPACE SHELTERS) -

Expansion Arm (Lazy Tongs)
Compact Capturing Device
Unfurlable Tape: STEM
Unfurlable Tape: Edge Cold-Welded
Unfurlable Tape: Edge Interlock (Zipper Boom)
Telescoping Shape

Inflated Shapes-Natural (Spiral Boom Toroids)
Inflated Shapes-Restrained (INFLATOMAT)
Resin and/or Foam Space Rigidized Shapes: Reinforced and Non-Reinforced
Inflated or Foam Filled Rigidized Toroid Stiffened Shapes

To appraise and compare these properly would require a comprehensive design and systems analysis. The concepts were therefore just briefly described in the Handbook and considered only with respect to specific mission requirements. A more thorough evaluation should consist of the following:

1. Comparison of various expandable structures concepts proposed for accomplishing similar functions. For example, the different approaches proposed for the design of solar collectors would be considered. These might include:
   a. Inflation/foam
   b. Mechanical/petal
   c. Mechanical/Fresnel
2. Comparison of the effectiveness with which the optimum expandable device and conventional devices fulfill the same mission requirements.

While structural efficiency is the primary criteria, other factors, including packaging ratio, cost, reliability, availability, etc., also deserve consideration in this analysis.

At this point, it should be noted that a realistic assessment of the packaging or expansion ratio must be made early in the design phase since a high ratio is not always necessary, or even desireable. For example, approximately one-third (1/3) of the enclosed volume of a space station is estimated to be occupied by "hard" equipment, and which would likely be held within the folded structure during launch. Thus, a ratio of 1:4 or 1:5 would be sufficient in this case.

MATERIALS - Expandable structures for aerospace applications depend upon the capability of materials to be packaged in a folded and/or compressed form, regardless of whether the material is rigid or flexible prior to deployment. In addition, such other desirable attributes present themselves as heat resistance, high strength, light weight, resistance to the space environment (e.g. radiation and vacuum), low cost, reliable functioning, easy fabrication, etc. The number of possible materials for a given application/device system therefore rapidly diminishes. These materials can be classified, as seen in Table 5 as either 'Primary' or 'Ancillary' (see Appendix), depending upon their relative importance to the basic expandable structure concept.

Table 5

AEROSPACE EXPANDABLE STRUCTURE
MATERIALS

PRIMARY	ANCILLARY
Metals	Sealants
Non-Metallics	Self-Sealants
Films	Adhesives
Elastomers	Lubricants
Resins	Coatings
Textiles	Thermal Protection Mat'ls
Coated Fabrics	Inflation Mat'ls
Foams	Rigidization Mat'ls
Composites	Semi-Permeable Membranes
Laminates	Radiation Absorbers
Sandwiches	

It is evident that the majority of the groupings are in the non-metallic family. On the other hand, most of the expandable structures sent into space, to date, have been essentially of metallic construction. This is not surprising when considered in the light of practical and realistic design considerations.

First of all, most designers, particularly in the aerospace industry, are intimately acquainted with metallic material properties and their conceptual usage. It is only natural that their flight hardware designs reflect this familiarity.

Secondly, the effects of the space environment on non-metallics are less readily understood as compared to metallics. This is especially true for the organic polymeric materials since (a) the number of possible and available combinations of chemical constituents is enormously greater for these materials than for the metallic ones, and (b) the various degradation phenomena are more diverse with the organics than with the metallics. For example, close approximation of metallic material loss in a hard vacuum can be calculated from the Knudsen-Langmuir relationship. On the other hand the equation is essentially useless with organic polymers since the vacuum degradation mechanisms are more complex (i.e. Molecular weight and crosslinking are functions of the composition proportions and the polymerization process utilized. Evaporation rate is also strongly influenced by variations in temperature and radiation damage).

Thirdly, the lack of combined space environmental design data has been especially noticeable with the organic materials since the mission operational environmental limits (particularly temperature) are beyond the known capability of many organic substances.

Finally, conventional design concepts for collapsible structures found suitable for many years on earth have been either of the type which could utilize atmospheric air for make-up pressurization, or were fabricated from mechanically foldable rigid elements. Concern over loss of pressurization media during space operation has slanted concepts toward the latter technique. Since the mechanical variety is readily designed with metallic materials and

whereas the organic polymers lend themselves to the typical inflated structures, the initial choice of material usage becomes self-evident. However, the trend is gradually shifting toward the non-metallics, particularly the organic polymers and ceramic textiles. This is due to several factors.

1. Non-metallic systems have greater potential for light weight, large packaging ratios and cheaper cost (all inherent in the use of elastomeric and rigid polymers and the organic and ceramic fibers).
2. Aerospace design engineers are being exposed to more non-metallic data, including space effects on polymeric materials.
3. Design techniques are being developed to protect susceptible organic materials to specific aspects of the space environment without sacrificing significant weight (e.g. vacuum deposited aluminum on mylar to protect the polymer from the solar radiation).
4. Repair and maintainability can be more readily carried out in space with the non-metallics, with greater safety and less launch cost. (e.g. self-sealants and space-cured laminates).
5. The increased efforts to develop more weight efficiency in space structures is accelerating conceptual sophistication of space deployment and rigidization techniques. This, in turn, is causing increased efforts to be placed in developing deployment/rigidization systems which are highly specific to space operations (e.g. plastic memory and vapor phase catalysis).

This trend is not to be misconstrued to mean that metals are disappearing from the expandable structure picture. In fact, similar design sophistication is likewise occurring in this area (e.g. unfurlable tapes which can be (a) self-rigidizing and retractable, (b) welded in space without heat and (c) rigidized via a "zipper"). Conventional metallic telescoping and hinged modules will likely be the first generation of manned orbital stations and laboratories. At the moment, metallic textiles, elastomer coated, appear to have the lead in the race for operational reentry decelerators.

Therefore, material data included in the Expandable Structures Design Handbook covers the gamit from metallics to non-metallics, but with particular emphasis on the latter. The quality of the data presented in the Handbook is, as originally anticipated, of a variable nature, depending upon test methods, facilities (specifically, space environmental simulation), quality and quantity of test specimens, and accuracy and completeness of the reported information. The so-called "space" environment data on materials, which is found in the literature, is of particular concern. Most of such testing has been carried out under far from simulated conditions. For example, available vacuum data thins out rapidly in the range of 10^{-5} to 10^{-6} torr, whereas typical manned orbital activities are contemplated for the 10^{-9} torr (300-500 nautical miles altitude) environment. Likewise, what little meteoroid testing which has been done, has been predominately carried out with particles traveling less than 25,000 ft/sec. Whereas actual near earth meteoroid velocities are a great deal higher than this.

Also, there is considerable difficulty in trying to compare radiation damage to materials for the following reasons:

1. There is a difference of opinion between radiation and material "authorities" as to whether radiation dosage is additive for all radiation, some types of radiation, or not at all. Radiation data has been generally obtained with cobalt 60 sources. The equivalency of this data to that which would result from actual space exposure is perhaps a gross assumption.
2. There is a distinct lack of uniformity in the test conditioning, especially when combined (i.e. radiation, vacuum, and heat) environments are involved.
3. There is very little consistency used to report the data; some "authorities" use rads, others use reps. Some report absorbed dose, absorbed dose rate, or even power. In addition, the data is liberally spread between c.g.s. and engineering units of about a dozen different combinations.
4. Previous polymeric and thermal history of some materials influence the radiation effects. Such data is usually unknown or unreported by the investigator. This could be the basis for some of the seeming discrepancies between reported data.

More recent but limited data is based upon a closer simulation of the space radiation. The lack of "space" data becomes even more pointed when combined environments (i.e. vacuum, radiation, and heat) are involved. Again, the principal limitation has been one of adequate test facilities. Further, such data that does exist usually merely involves the exposure of the material to the combined environment, but doesn't simultaneously test the properties (e.g. tensile strength). Lubricants and bearings appear to be the only types of material which have been functionally tested to a significant extent in a space environment, and the results are presented in the applicable sections of the Handbook.

Since expandable structures utilize materials in "unconventional" design or utilize unconventional materials or material combinations in conventional designs, it is extremely important that adequate testing substantiate the choices for the particular mission. Unfortunately, the testing is carried out in the mission itself and not prior to launch. The lack of testing is generally ascribed to (a) shortage of adequate facilities to properly ascertain the functioning reliability of the material, (b) the lack of sufficient time to conduct such testing prior to "freezing" of the design, and (c) the often exorbitant cost in conducting sufficient testing to satisfactorily define the characteristics of the material in the specific environment to be encountered.

Technical progress in the development of new and/or improved materials will permit a more effective choice to be made for a given mission and function, but the ability to make the best compromise between allowable design parameters and materials properties depends upon the knowledge of the relative importance of each factor entering into the design. This is partially due to the interdependence of many material properties and the feasibility of shifting a given material's properties by modification of fabrication techniques (e.g. heat treat temperatures for metals and curing pressure of laminates).

It is evident that aerospace expandable structures will involve a broad spectrum of material needs. Some of these are being met today, others will require improvements. In some cases, these improvements merely depend upon sufficient budget to develop the data. In others, a scientific or technical "break-through" is indicated. Our investigation of the expandable structures state of the art pointed-up the fact that these material limitations were usually the determining factors in distinguishing between so-called first and second generation expandable structure concepts. Our analysis also resulted in the following conclusions and recommendations concerning the specific types of materials of most interest.

Primary Materials:

Metals - There has been insufficient data reported on the effects of cold-work history on the mechanical properties of thin foils, metallic films, and fibers. It is known that highly cold-worked material exposed to both low pressure and radiation can evaporate preferentially at high energy sites, i.e. sites of dislocation, stack-up, or grain boundaries, at much lower temperatures than predicted by the Langmuir equation. While the rates would probably be low, this preferential attack could not only affect the optical properties of the surface, but may act as significant stress concentration factors. Therefore, an investigation into the quantitative behavior of these materials under ambient as well as space conditions, appears in order so that design allowables can be established.

Although, by proper choice of metals, in-space utilization for expandable structures has been found generally satisfactory, a significant amount of development is still required for the reentry mission. For such application, present design efforts are limited by a lack of design allowables for metal fibers and fabrics, particularly in biaxial stress and fatigue. Due to variations in crimp, twist, and tension created during the various plying and weaving operations, even simple tensile properties of the fabrics are not sufficiently known, especially in a high thermal input environment. It is also necessary that better correlation between the properties of the fibers and of the woven textiles be established. This must include the determination of the effect of weave upon mechanical (e.g. biaxial) strength.

The present operational thermal capabilities of these thin fibers are limited to 1300-1500°F.; some expandable earth orbital reentry configurations have a potential of 1800-2200°F. To date, the high temperature refractory alloys have been hindered by a lack of adequate oxidation protection. In order to achieve flexibility in the fabric, thinner fibers may be required. Unfortunately, the effect of oxidation then becomes more pronounced due to the decrease in crossectional area. The development of oxidation prevention coatings has been initiated via several ASD contracts, but considerable effort still appears required before a reliable system will be made available.

Preliminary data indicates that the comparatively rigid aluminides, beryllides, and silicides, and elastomeric organo-silicone coatings are among the more likely candidates.

Air permeability and shock-loading of these "reentry" fabrics have received only minimal attention; most of it being theoretical, very little experimental. However, this is somewhat understandable since very little of such fabrics have been made available to date.

Films - This category of materials provides the lightest weight and most foldable expandable structures. However, other than H-Film, the typical ones (e.g. Mylar, Tedlar, Aclar, and Saran) considered for this use, are significantly sensitive to the combined effects of natural space radiation and vacuum. Scattered data indicates that vacuum deposition of metallic coatings alleviates the radiation degradation. Since the stability of these films in vacuum, alone, is quite good, temperature then becomes the determining factor. Here too, the new H-Film appears to be better suited for usage in space than the other films due to its better retention of mechanical and physical properties at high temperatures. For example, the Zero Strength Temperature (20 psi load, 5 sec. to failure) for H-Film is reported to be 800°C, as compared to 230°C for Mylar. However, this film is not heat-sealable. An adhesive is needed which will result in bond strengths comparable to the film strength, have the radiation and thermal stability of H-Film, and add little to total weight and fabrication cost.

Elastomers - Butyl and polysulfide elastomers must be protected from the space environment. Natural rubber is limited to non-ozone containing atmospheres. Other than silicone elastomers, the others, including the fluorine containing types are temperature limited, especially if subjected to abnormal doses of nuclear radiation. Requirements for flexibility below -70°F are limited to the use of silicone elastomers. Use of short chain elastomers or inclusion of plasticizers in elastomers are not recommended for space applications due to the higher volatility of these materials. It is apparent that elastomers must be chosen with considerable care when used in expandable structure systems, even though they are not normally considered the load bearing member of the structure (i.e. fibrous reinforcements are added to carry the loads). Since the primary functions of the elastomers are to control permeability and provide oxidation protection to metallic fibrous reinforcements, design data for these characteristics are needed. Some of this information is presently available on the base stock, and can be used for comparative purposes, but design allowables can not be extrapolated from these values; the actual coated fabric under consideration must be tested. There is presently insufficient data of this type.

Since the thermal properties of even silicone elastomers are quite marginal, the development of better elastomers for extreme thermal and radiation environments are considered necessary.

Textiles - This class of materials is certainly as important to expandable structures as any other material class, if not the most important. It can be used alone (parachutes), coated with rigid or elastomeric polymers, sandwiched with films, or laminated into a multi-ply structure. The continuous filament fibers are generally preferred to the staple variety for space systems because of their higher strengths. These fibers are used individually or as woven goods.

The worst textile problems occur during reentry; primarily due to aerodynamic heating. The commercially available organic fibers, with temperature limits of 500°F (50% strength retention of 0.07 mil diameter HT-1 fibers after 10 minute exposure) are of no use at reentry temperatures. Although experimental runs of polybenzimidazole (PBI) fibers show promise of an added 100-150°F, the principal attack on the high thermal regime is being made with

ceramic fibers, and the afore-mentioned superalloy and refractory metal fibers. The problems involved with the metal fibers have been mentioned. The limitations to the use of high strength and high modulus ceramic fibers (fiberglass, leached fiberglass, and quartz) are principally based upon their inherent brittleness, complicating handling and manufacturing, and difficulty to bond with organic polymers, thereby preventing efficient load transference from the weaker and more elastic matrix materials.

There should be a continuation of the broad industry and ASD approach to resolving the different levels and combinations of aerospace textile requirements, especially in the areas of continuous ceramic oxide filament forming and high modulus fiber flexibility. The development of adequate manufacturing techniques and/or surface finishes for quartz should be pursued so that mechanical properties close to theoretical, can be achieved.

Whereas the basic organic fibers are limited to temperatures below 1000°F, the pyrolized or fired organic fibers can be used to at least 5500°F, depending upon the degree of conversion to pure carbon. Although their strengths are less than that of their precursor organic fiber, their strength retention at reentry temperatures is excellent. However, as in the case of refractory metals, the main difficulty in their use as reentry textiles is due to their rapid degradation in the oxidizing atmosphere present during reentry. Preliminary,but successful,attempts to resolve this problem by converting the fibrous carbon to high temperature carbides and/or coating the carbon fibers with transition group metals, have been reported. The next phase should be their evaluation under reentry conditions. Depending upon strength and fabricability, such materials could represent a significant break-through in collapsible reentry decellerators.

Elastomer Coated Fabrics - This type of construction is the basis of most structural load bearing, pressurized expandable structures. As previously indicated, the elastomer provides a permeability control and also protects the fibrous substrate from the initial impact of the surrounding environment, including the ultraviolet and vacuum of space. The primary loads are assumed to be carried by the textile. However, in most cases the surface temperature will dictate the choice of fabric as well as the coating.

Several elastomers and glass-frit loaded elastomers are being evaluated as coatings for reentry metallic fabrics. The loaded system is apparently more effective than the elastomer alone. But its operational reliability is more questionable since it is dependent upon careful sequential timing and thermal input during the reentry cycle. It is particularly critical between the time/temperature moment when the elastomer matrix significantly degrades, and the moment when the frit vitrifies and coalesces sufficiently to be retained on the fabric. Since this technique is so highly sensitive to the reentry time/temperature profile, it will be necessary to carry out final proof testing in space. However, prior to this flight testing, formulation improvements must be sought, and considerable testing carried out to establish the parametric time/temperature values and their attendant realistic tolerances.

Resins - Expandable structure usage of resins range from teflon and nylon bearings to space rigidized epoxy laminates. Space environmental data is lacking on the thermosetting resins by themselves; somewhat more data is available on

thermosetting resin/fiberglass laminates and the thermoplastic resins. Functional testing of nylon and teflon has also been carried out in simulated space environments. In general, it has been found that the fully polymerized resins are quite stable in vacuum alone, but degrade at various rates with increasing radiation and temperature.

At this point it would be quite simple and expected to express a desire for "better" (stronger and more space-stable) resins. But, in examining the realistic resin needs for expandable structures and comparing them to the available types and sub-types, we are confronted with the fact that the off-the-shelf resin systems have only been barely exploited for expandable structures. For example, space rigidization via vapor catalization has been examined primarily with respect to epoxy and urethane systems. Does this mean that the oxygen cured polyesters are not usable? Can we likewise stipulate at this time that PVC is the best resin to use in rigidizing via plasticizer volatilization? Obviously not. The number of alternate, and possibly better, resin choices for the space converted systems are particularly susceptible to further examination.

As previously noted, the organic polymers are not suitable for reentry systems (except as ablation materials). Since in-space structure temperatures will be somewhere between -150° to 300°F, depending upon the absorptivity/emissivity ratio of the external surface of the structure, there is presently no shortage of resin systems which can be used (temperature stability being the criteria).

On the other hand, reentry conditions point up an interest in maintaining the investigation of new organo-inorganic and chelating polymers at a respectible level, until a useful and practical system is developed. At such time, the effort should be accelerated.

A particularly interesting use of resins (in the forms of films or coatings) for expandable structures is based upon techniques which will cause the resin to degrade to the point of being readily dissapated in the space environment. These space degradable materials offer the opportunity of developing an initial α/ϵ different than that required for the operational thermal balance of space structures. For example, a high temperature surface would be needed for many of the chemical and physical rigidization processes being considered. However, such temperatures (i.e. 250-350°F) are considerably higher than that permitted for manned operations. Therefore, it is necessary that the initial surface characteristics which can develope the above high temperatures for rigidization, be removed or modified so that the lower operational temperatures can be obtained. A number of possible techniques which might be utilized to create such a self-correctable surface, present themselves and should be investigated. The space degradable type, which can have suitable pigmenting characteristics itself, can, in turn, be broken down to several different possible mechanisms:

A. Polymeric materials which are readily vaporized at the low pressures available in space but which are sufficiently stable during the initial rigidization operation.

B. Thermal degradation of the polymer to the point of obtaining chain scission, creating lower molecular weight fractions which are readily vaporized.

C. Activation of the polymeric species to form an increased number of selectively created free-radicals which do not have a high probability of cross-linking due to a more stable configuration of the polymer (e.g. steric hindrance). This might be accomplished by radiation and/or heat alone, or in conjunction with selectively chosen free-radical activating initiators.

Other techniques might be any combination of those mentioned above or possibly completely different approaches.

Although such materials have an obvious usefulness for "in-space" processing, no significant activity along these lines were detected during the Survey.

Foams - Although the foams herein considered are based upon organic polymers, we classify them separately from the Resins and Elastomers groups since they very specifically affect a large number of expandable structure configuration concepts. Most of the "earth" foamed systems (epoxies, phenolics, silicones, polyurethanes, etc.) could be "expanded' in space as panels. "In-space" foaming has been generally limited to polyurethanes. Solar collectors, meteoroid barriers, and thermal insulation are presently attracting most of the "in-space" foaming interest. Although quite a variety of techniques for foaming "in-space" are being evaluated, very little quantitative comparison has been carried out, either with respect to each other, or in comparison to similar foams prepared under "earth" conditions. Until such data is obtained, realistic design allowables can not be established. This will necessitate the use of very conservative values in designating the strength or stiffness/weight ratio of the system.

Laminates and Sandwiches - These two classes of materials provide higher strength and higher modulus to weight ratios than that which can be obtained with the materials previously indicated. They can be fabricated on earth or in space. They offer the only serious potential competition to the various conventional metallic structures being considered for man-in-space operations, particularly the orbital vehicle types.

As on earth, the optimum material combination for such application would be a rigid resin/filament wound structure. The feasibility of deploying and rigidizing such a system in even a simulated space environment has yet to be demonstrated. But this approach is especially attractive since the materials are basically off-the-shelf; the principal development effort being in the folding, packaging, and deployment of the lay-up without causing fiber distortion. This will involve optimization of tackiness, viscosity, barrier films, etc; all of which are apparently within the present state of the art. The feasibility of this concept should be verified as soon as possible so that it can be given realistic consideration in our present space ventures; a small delay now could easily eliminate it from space designs for quite some time since first generation manned orbital modules are being designed now. An evaluation of such scope should include the comparison of the mechanical properties of the space rigidized system with those of an identical terrestrial rigidized one.

Similar comparisons should also be carried out with the more conventional laminate and sandwich lay-ups. An evaluation with respect to generally accepted conservative design allowables, such as specified in MIL-HDBK-17 and MIL-HKBK-23, respectively, should be included.

Ancillary Materials:

Sealants - Since sealants are normally protected from the space environment by the adjacent structure (except at the edges), most commercial formulations will have some applicability. Therefore, only very limited sealant research has been carried out specifically for space usage. Other than silicone RTV types, most room temperature curing systems will be limited to -75° to 275°F. The silicones have a useful range of -100° to 500°F.

With regard to expandable structures, low priority is indicated for the development and evaluation of improved high strength, room temperature curing sealants with wide temperature stability. These could include polyester and fluorinated types. On the other hand, the development of better and simpler systems for in-space repair should be accelerated.

Self-Sealants - The mechanical types appear to be the most useful at this time. These include compacted closed cell flexible polyurethane foams, restricted thin film micro-teflons, and/or retained laminated rubber fabric elastomeric sealant combinations. There is a distinct lack of design data, as well as feasibility or workability information regarding other self-sealing concepts such as chemical reactant sealants, latex sealants, and swelling sealants.

Adhesives - "Earth" cured structural adhesives, conforming to MIL-A-5092D, are useful for expandable structures within the same temperature limitations that they have on earth. There is a need for better room temperature curing adhesives which offer the high strength properties presently found only with the heat cured systems. As noted previously, bonding of H-Film is still a problem. The use of presently available 100% solids structural adhesives appears feasible for in-space fabrication or repairs of semi-structural and non-structural assemblies, but the actual techniques have as yet to be defined. Solvent release adhesive systems are not considered useful for in-space bonding due to the difficulty of controlling the dissipation of the solvents, and the shrinkage usually encountered with this type system as the solvents are removed.

"In-space" adhesive bonded joints should be evaluated in comparison with "earth" fabricated ones.

Lubricants - The principal use of lubricants for expandable structures involves the hinged, geared, or telescoping systems. Unless lubricants are used, cold-welding of these structures can occur. Lubricants currently used or proposed for use on first generation expandable structures appear to be suitable for second generation structures. No technological breakthroughs seem necessary, although some design problems still exist. Extensive information is available on the properties of most lubricants, but considerable disagreement exist on space environmental effects upon nylon and teflon.

The particular choices of lubricant for expandable structures should be verified by tests simulating the actual service conditions as closely as possible. Only minor testing has been carried out on lubricants operating under high torque combined with high rpm while exposed to the space environment.

Coatings - Fabric permeability control is the most important use of coatings for expandable structures. Passive thermal control is next in significance. The protection of organic polymers from radiation degradation and the protection of fibers from oxidation complete the areas of usage. The same general types of thermal control coatings will probably be used for second as well as first generation systems. Organic base paints are being primarily used as thermal control coatings in space instead of the more stable inorganic ones due to the ease of application and low cost of the former type.

Self-thermostatic coatings, with their α/ϵ ratio changing with the thermal input, could minimize temperature fluctuations of space structures. Very little information on these special coatings appears in the literature. A systematic survey of industrial and government activities to develop self-thermostatic coatings, should be conducted. Based upon the results of such a survey, an optimization program should be initiated to develop and evaluate such materials for use on expandable structures (and other aerospace devices). This subject is further discussed under Radiation Absorbers.

Thermal Protection Materials - In addition to their use as temperature controlling systems for deployed structures, these materials can also perform specific critical operational tasks in certain expandable structures designs (e.g. thermal control of chemical or physical rigidization processes and provide part of the foldable envelope in the MOOSE personnel recovery concept).

Conventional insulation and passive thermal control coatings appear satisfactory for most in-space operations. However, as could be expected, reentry presents a considerable challenge in this category. From the standpoint of simplicity, the flexible ablators are of the most interest. Unfortunately, very little published literature was found concerning them; none of it being design data. Extensive work will be required to obtain such information. The importance of the task warrants early action.

Inflation Materials - Inflation or pressurization materials, other than the well-known bottled gasses, can provide the necessary deployment of many types of expandable structures, including those which are rigidized by chemical techniques in addition to those which are stabilized by the self-same deployment pressure. Our investigation elicited the identification and quantities of some of the subliming materials used (e.g., benzoic acid, anthroquinone, etc.) However, the data was considered insufficient to warrant its inclusion in the Handbook at this time. A specific survey of the literature and pertinent organizations utilizing these inflation materials, should be conducted in order to more closely define the state-of-the-art.

Rigidization Materials - Many of the chemical and physical rigidization processes applicable to expandable space structures utilize activating or inhibiting agents. These include such off-the-shelf items as blowing agents

and peroxide catalysts. Molecular sieves, being newer, offer less variety, but can often be tailored to specific requirements. In varying degrees, these materials represent weight penalties and contamination in the system.

It is suggested that data be obtained, on a low priority basis, indicating the relative merits of typical commercial blowing agents, when used with standard resins in a typical space environment. On a similar schedule, the space rigidization effectiveness of peroxide catalysts in conjunction with various polyesters (i.e., thermal and ultraviolet activated latent premixed systems, and vapor catalization) should be determined.

Radiation Absorbers - This class of materials is included because of its effect upon thermal control coatings, films, laminates, elastomers, etc. Some ultraviolet absorbers effective on earth are not necessarily usable for space applications due to their high volatility in a vacuum. For example, benzophenone and benzotriazole types are much less effective in space, whereas the benzoyl substituted ferrocene compounds maintain full capability, especially when exposed to ultraviolet. Long term usage in space further complicates the choice since most radiation absorbers gradually weaken with time, cycling, and intensity of incident radiation. The data in the literature indicates a sporadic, and often qualitative, examination of these variables and their effect upon expandable structure materials in a space environment. A more definitive approach is required. For example, a concerted effort should be made to develop such information on typical expandable structure polymers in the operational forms (e.g. films, coatings, laminates, etc.), and which contain the more useful anti-rads, permitting a comparison with the "normal" polymer formulations. Thus, actual design data can be obtained rather than merely indications of which anti-rads are better than others.

Although not actually radiation absorbers, the thermotropic/photochromic materials represent an almost untapped field of thermal and/or radiation control, particularly the reversable types. This results from the condition that, in general, thermal control coatings which have been applied to aerospace structures, are basically of a constant α/ϵ ratio (except as gradual changes occur with time due to degradation of some of the vehicles and pigments used). A thermostatic or self-regulating thermal coating, having a variable ratio, could effect a stabilization of the temperature regardless of the motion of the structure with respect to such energy sources as the sun. During the Survey, it became apparent that very few organizations were actively researching this problem. Two basic approaches to this concept seemed to cover what little information was revealed. One type involves chemical compounds which have variable transmission characteristics, depending upon irradiation frequency and intensity. The second type is based upon the use of immiscible materials which are "cloudy" until irradiated or heated, whereupon the solubility increases and the material mixture becomes transparent. Interestingly, the first type particularly, suffer the same fatigue, evidenced by lack of longevity, as typical radiation absorbers.

Permeability Membranes - Membranes having known and controlled permeability to specific vapors are required for certain chemical and physical rigidization techniques. Very little data has been collected and collated on this subject. A survey/literature search pertaining to it would minimize duplication of evaluations and indicate probable membrane systems for a given application.

PROCESSING - Similar to conventional aerospace structures, the expandable types have been, and will be, essentially fabricated on earth. However, expandables as a class, involve more in-space operations than others, especially with regard to their deployment and rigidization. These expandable structure in-space, non-functional operations include:

1. Assembly of preformed components (furniture, partitions, etc.).
2. Fabrication of structures (chemical or physical rigidization).
3. Modification of structure (addition of "hard-points" and modules, porting, etc.).
4. Repair of damaged structure (meteoroid penetration).
5. Maintenance of structure (lubrication, resealing, recoating, etc.).

As can be expected, these in-space operations are considerably more limited in scope than comparable ones carried out on earth. For example, adhesive bonding with presently available adhesive systems to meet the mechanical property requirements of MIL-A-5090D or MIL-A-25463, necessitates certain minimum substrate surface preparations and specific cure cycles. Assuming that the surfaces could be prepared on "earth" and maintained in the desired condition until bonded in space, the need for high pressures and high temperatures for the cure cycle would, at present, prevent the attainment of the desired bond properties. A similar situation exists with regard to some laminating and sealing systems. The development of in-space curing techniques for polymeric systems requiring high temperatures and/or high pressures does not appear to involve technical "break-throughs", but the properties obtained must be evaluated in terms of the cost, launch weight of facilities, in-space processing manhours, etc. At present, neither experimental nor analytical investigations have been carried out to ascertain the adviseability or even feasability of conducting many of these in-space processes. For example, various sealing techniques have been developed for "earth" processing. However, similar operations in space may require sealant reformulation, or modification (or even complete redesign) of application equipment.

Similar problems also occur in the "in-space" coating, bonding, or laminating processes. Thus, the applicable formulations would involve such considerations as packaging, types and amounts of solvents, viscosity, tackiness, storage and "pot" life, etc. Considerable effort appears warranted in these "in-space" processing areas, especially for long term space missions.

The "in-space" processes of deployment and rigidization are, obviously, unique to expandable structures. However, some of the methods utilized are closely akin to "earth" techniques, mechanical deployment via hinges being typical. On the other hand, the chemical and physical rigidization methods are generally new or extensively modified "earth" systems. Therefore, design data for such methods is, at best, sparse. For example, deployment and rigidization via "plastic memory" is potentially an important expandable structure process due to its inherent simplicity. But published information of design value only covers the use of polyethylene. What of other polymers, such as polypropylene, which have better thermal and mechanical properties?

As previously noted, in-space foaming is receiving considerable attention for such applications as solar collectors and meteoroid barriers. The principal techniques under consideration include separately contained liquids

(double tank arrangement), encapsulated liquid reactants mixed with solid co-reactant, and the completely solid reactants. These processes are susceptible to considerable variation in quality of product due to number of comparatively uncontrolled, or difficult to control, variables when carried out in a space environment. To date, only limited success has been reported in foaming at 10^{-6} torr; no data has been cited between 10^{-6} - 10^{-9} torr. Another problem area includes the difficulty of foaming in thin sections. In general, the data has only involved feasability, not design allowables.

Likewise, what little investigations of "in-space" molding of laminates and sandwiches which have been or are being conducted, involve only feasability. Preliminary data should be obtained on such structural panels and compared with similar panels fabricated in an "earth" environment. It is also considered adviseable that accelerated investigations be carried out to determine the fabrication variables, and their importance, involved in the overall area of space rigidization systems utilizing polymer/reinforcement combinations, with particular emphasis being given to the impregnation, handling, packaging, storage, etc., of the combined systems.

Expandable structures present somewhat unique "timing" problems with respect to many of these processes. Thus, certain chemical rigidization techniques require excessive cure times, delaying full utilization of the structure. Likewise, volatilization processes dependent solely upon the vacuum of space would be limited to missions involving sufficient time to carry out the process. Design data for this process, including such information as strength versus time, is presently insufficient.

Concern with process time also refers to the initiation of the process. For example, "instant" initiation, upon command, of space polymerization methods is considered highly desireable; its attainment is usually extremely difficult. An examination of possible techniques might include such triggering devices as electrically conductive wires, fibers or coatings, pyrotechnics, encapsulated reactants (encapsulating coating removed upon command), etc. Their evaluation should include such factors as weight penalties, reliability, and predictable time tolerances.

The above comments are not intended to infer that the only expandable structure processing problems occur in space. Actually, some very basic material restrictions exist due to "earth" processing limitations. Some typical examples can be cited:

1. Present forming of, and weaving techniques for high strength, high temperature metallic fibers are far from satisfactory; design allowables are non-existant. These are particularly critical items with regard to reentry recovery.
2. Handling and fabrication variables of self-abrading, high temperature woven textiles (e.g. quartz and leached fiberglass) should be investigated to reduce loss in basic fiber strength due to these factors.

SUMMARY

In summary, it can be stated that the growing interest in expandable structures is a consequence not only of significant volume and weight reductions attendant to these concepts, but also of the unique configuration possibilities inherent in deployment and rigidization in the space environment. However, it is evident that serious questions are in order concerning the present feasibility of some of the expandable structures concepts being propounded, due not only to the state-of-the-art of material development and testing but also due to the lack of a thorough knowledge of the environment to be encountered in space and its effect upon materials. It is clear that there is a distinct difference between a logical concept/material combination and a feasible one.

It has also become increasingly apparent to those of us involved in the Study that a closer liaison between the designers and the materials engineers is going to be essential to the usage of expandable structures, particularly for reentry applications. Many of the trajectories and design concepts involve the utilization of materials in constructions and environments heretofore not considered in "classical" aerospace engineering. In order to expedite the progress in this area, it is necessary that the material requirements be clearly spelled out for a particular mission and configuration. The optimum choice of materials may be available off-the-shelf. On the other hand, a technical "break-through" may be required. We have, therefore, reviewed in this paper some of the problem areas pertainent to the design of aerospace expandable structures, with particular regard to the materials utilized in such designs. Some of these problem areas are presently under examination by ASD and other interested government agencies, as well as by various industrial organizations; some are merely in the "discussion" stage. The utilization of particular expandable structures for aerospace applications will be contingent upon the vigour applied to many of these areas of unknown feasability and design data.

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APPENDIX A

- DEFINITIONS -

<u>ANCILLARY MATERIALS</u>
Those materials which are subordinate to the basic design, but can be essential to the operation of the structure.

<u>DEPLOYMENT</u>
The technique which permits the expansion, unfurling, extension, or erection of a structure from a compacted packaging arrangement to one involving a larger dimension or enclosing a greater volume of space.

<u>PRIMARY MATERIALS</u>
Those materials which are fundamental to the basic concept or design of the expandable structure device.

<u>RIGIDIZATION</u>
The technique which permits a deployed structure to maintain its shape against collapse or deformation.

ON THE STRUCTURAL BEHAVIOR OF INFLATED FABRIC CYLINDERS UNDER VARIOUS LOADING CONDITIONS

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The use of rigid structures for space vehicle applications is limited by the load and volume requirements imposed on the structure during the boost condition. The fixed volume of a rigid structure contributes to aerodynamic drag and the structure must be designed to the high loads and temperatures occurring during boost. The rigid structure may be considerably overdesigned for the loads and temperatures encountered in space. Consequently, a large amount of interest has developed in using inflatable structures for space and reentry vehicle applications.

In developing structural analysis methods for inflatable structures, it is found that classical methods long used for rigid structures do not apply. The material of which an inflatable structure ordinarily is fabricated is neither isotropic nor orthotropic in behavior, but is anisotropic. Furthermore, when non-metallic materials are used, the large deflection behavior of the structure must be accounted for as a result of large strains at the design stress level.

This paper introduces an analytical method for predicting structural behavior of inflatable fabric cylinders. The behavior is dependent on the anisotropic and the large deflection stress-strain characteristics of the fabric material.

PROPERTIES OF MATERIAL

The structural behavior of inflated fabric cylinders depends upon the properties of the fabric material such as anisotropy and the moduli of elasticity. A typical fabric material usually consists of fibers that are placed in two (or more) directions and impregnating resin. The fibers in one direction are not necessarily of the same spacing or material as the fibers in the orthogonal direction. Two moduli of elasticity are usually defined; one for the direction of each of the sets of fibers. For any direction

which is not parallel with one of fiber directions, the effect of modulus of elasticity of the fibers is negligible since the system of fibers becomes essentially a mechanism. A third characteristic modulus is identified with the resin. The modulus of the resin is, in general, much smaller than the moduli of elasticity of the fibers. The impregnation material can be regarded as orthotropic or even isotropic.

Since all layers of fibers and impregnating resin work together without separation, the same elongation must occur for each element of the composite. It is necessary to work with the anisotropic modulus of elasticity, which varies with the angle of inclination to the main directions of fibers, as illustrated in Figure 1.

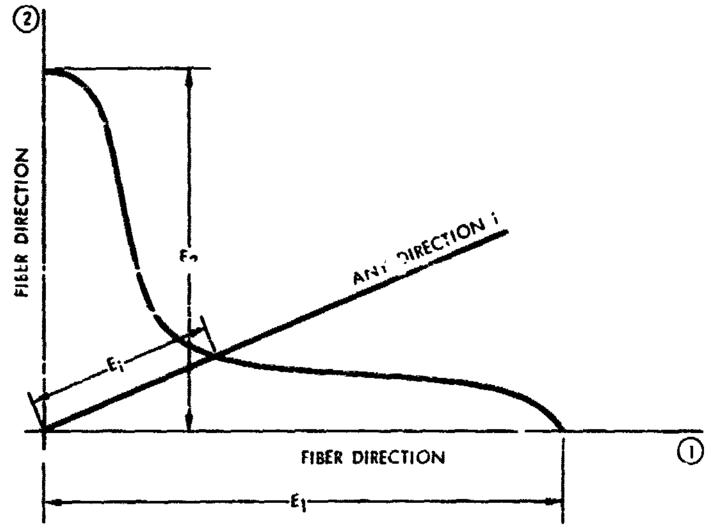

Figure 1. Modulus of Elasticity (E_i) Versus Fiber Direction

For a principal direction along a set of fibers a simple approximate formula can be derived:

$$_iE_{eff} = E'_i \eta'_i + E_i \eta''_i$$

where

$_i$ = Subscript which indicates the direction of the fibers

$_iE_{eff}$ = Final, combined modulus of elasticity in i direction

E'_i = Modulus of elasticity of fiber material

E''_i = Modulus of elasticity for impregnation

η'_i = Concentration of fibers = $\frac{A_f}{A}$

A_f = Total area of fibers

A = Total area of fibers + impregnation

η''_i = Concentration of impregnation = A_r/A

A_r = Area of resin.

η'_i and η''_i usually are known for the fabric. For example, the following data are tabulated for 2 ply, 4.8 oz., 2.5 mil thick neoprene.

Table 1. Material Characteristics

Direction	Combined $_iE_{eff}$	Concentration η	Moduli of Elasticity lbs/in^2	
			Fibers only	Impregnation
Fill	35900	0.0426	840000	4000
Warp	55700	0.0522	840000	4000

The approximate formula $_iE_{eff}$ can be checked using the above data.

Fill direction:

$$\begin{aligned} {}_1E_{eff} &= 840000 \times 0.0426 = 35800 \\ &\quad 4000 \times 0.9574 = \underline{\ 3820\ } \\ &\qquad\qquad 39620\ lbs/in^2 \end{aligned}$$

≈ 35900

(Difference 10%)

Warp direction:

$$\begin{aligned} {}_2E_{eff} &= 840000 \times 0.0522 = 44000 \\ &\quad 4000 \times 0.9478 = \underline{\ 3800\ } \\ &\qquad\qquad 47800\ lb/in^2 \end{aligned}$$

≈ 5570

(Difference 16%)

The results are satisfactory in view of the uncertainty in the value of E which varies with the material and may even vary within a given section of material.

It may be concluded that if the direction of fibers is the same as the direction of the principal stresses, the values for E for the fill and the warp directions should be determined from uniaxial tests. These values, however, will be changed considerably if the fibers are placed at some angle to the directions of principal stresses. The data for the latter case should be obtained in a test of an inflated fabric cylinder for which the fibers are placed at some angle to the longitudinal and circumferential directions. This case is not usually encountered with cylinders, but it is very important for the inflated cone where all fibers cannot be in the circumferential and longitudinal direction.

The behavior of inflated structures is further complicated even for one chosen direction, since the modulus of elasticity varies with the applied load. This variation is due to two causes: a) the actual behavior of the material of the fibers, and b) the tendency of fiber to straighten under load. (The fibers are initially curved because of the cross-directional orientation).

The shear modulus, G, is a function of a) impregnation material, b) tension in fibers, c) concentration of fibers, and d) friction between the fibers. The following formula has been found to reasonably approximate G (for neoprene material):

$$G \approx 1910 + (5.7 - 0.16p)\,\sigma_f \quad \left(\text{lbs/in.}^2\right)$$

where

G = Shear modulus

p = Internal pressure (lbs/in.^{2})

σ_f = Stress in fibers.

This formula has been verified by test and is considered to be a reasonable approximation for neoprene if permanent set does not occur. The history of loading influences the material behavior very much, as is shown in the discussion on torsional loads.

ULTIMATE BIAXIAL STRESSES

It has been observed in tests that a fabric loaded biaxially will fail at a lower value of tension than in a uniaxial test. Therefore, it is very important to be able to determine the burst strength under internal pressure.

Uniaxial tests performed at S&ID on 2 ply, 4.6 oz, 2.5 mil. neoprene gave the following data.

Table 2. Uniaxial Test Results

Direction	Elongation	Critical Tensional Stress
Warp	19.6%	709 lbs/in.
Fill	35.4%	454 lbs/in.

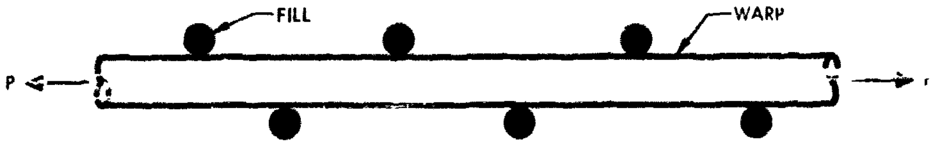

Figure 2. Fiber Under Uniaxial Test

Referring to Figure 2, assume that a uniaxial test is made in the direction of warp. The fibers will be elongated as the specimen is loaded. Also assume that a uniaxial test is made in the fill direction on a strip as is shown in Figure 3a and b.

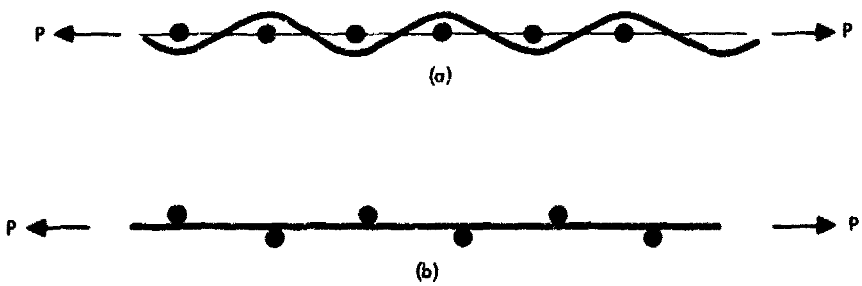

Figure 3. Uniaxial Test in Fill Direction

The same load will cause much more elongation in the uniaxial test because the fibers must be straightened before significant elongation can occur. This explains why the uniaxial test probably shows such a difference in elongations in the warp and the fill directions. The different stress can probably be explained by the different concentration of fibers in the fill and the warp direction since the material of the fibers is the same.

Such results are not observed in a biaxial test since both sets of fibers are stressed simultaneously and neither set of fibers can assume a straight line position. This effect may also introduce, in some cases, a shortening of the original length of the test cylinder. This effect was observed in pressurization tests of some fabric cylinders when a shortening in the longitudinal direction rather than an elongation was noted. This effect was evident when the warp was placed in the longitudinal direction.

It is well known that a rod loaded simultaneously with an axial force and a shear as per Figure 4 will fail at a lower load than if the same rod is loaded with axial load only. The failure may occur either in tension or in shear, depending upon the loads and the material.

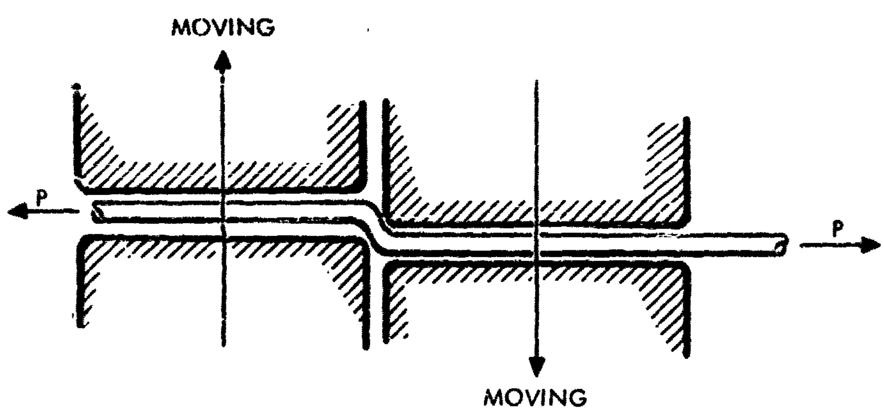

Figure 4. Rod Loaded With an Axial Force and Shear

The limitation of the allowable load can be established with the interaction curve as per Figure 5, where

σ = Tensile Stress

σ_I = Ultimate tensional stress for case $\tau = o$

τ = Shear stress

τ_I = Ultimate shear stress for the case $\sigma = 0$

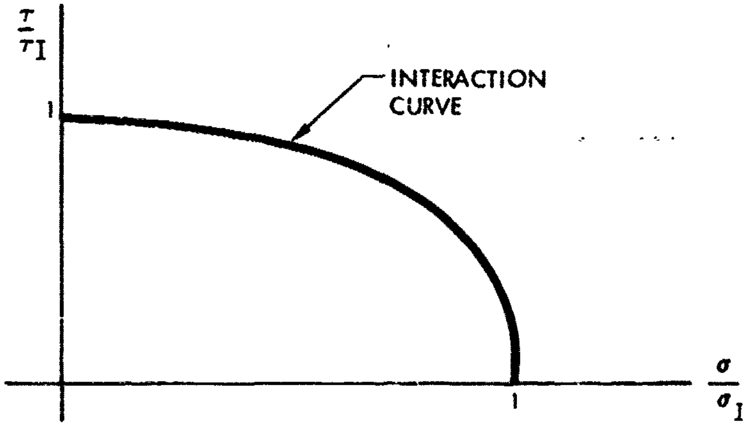

Figure 5. Interaction Curve

Since the fabric is stressed in two perpendicular directions, the fibers in both directions will assume some deformed shape as per Figure 6.

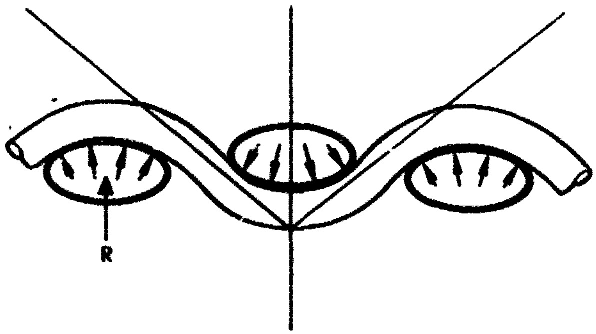

Figure 6. Loading on Deformed Fiber

The fibers will stay deformed and the tensile force causes each fiber to exert a pressure, R, on an orthogonally placed fiber. The loading, R, will induce a shear as is shown on Figure 7.

In general, the cross section of the fiber is unknown, but it can be approximated by assuming that a uniform pressure results from R as is shown on Figure 6. The distribution of the fibers is unknown, too; however, some ideal shape and distribution may be assumed.

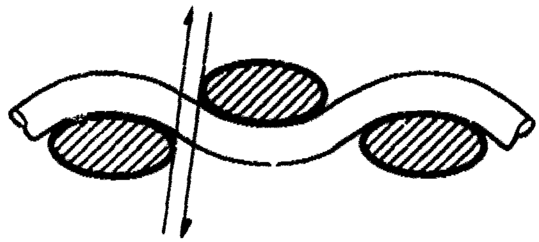

Figure 7. Shear Load on Deformed Fiber

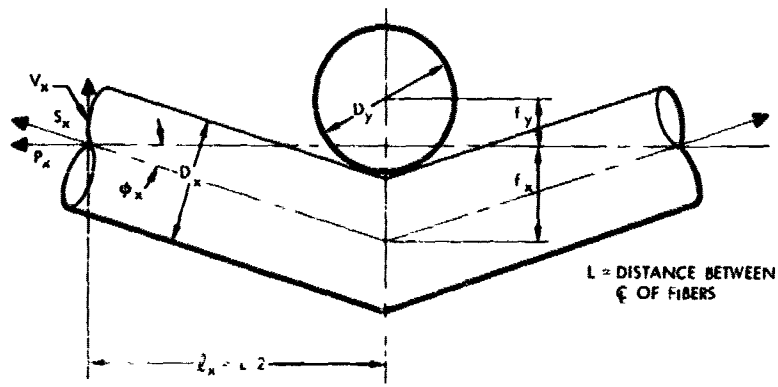

Figure 8. Geometry of Deformed Fiber

In accordance with Figure 8 the following formula for shear can be derived:

$$V_x = 1/2 \, \frac{P_x P_y}{P_x \ell_y + P_y \ell_x} \, (D_x + D_y) \tag{1}$$

For the special case in which

$$D_x = D_y = D \tag{1a}$$

$$\ell_x = \ell_y = \ell$$

$$V_x = \frac{P_x P_y}{P_x + P_y} \, \frac{D}{\ell}$$

In using the above formulas it is necessary to know D_x, D_y, ℓ_x, and ℓ_y where ℓ_x, and ℓ_y are half of the distance between the center lines of the fibers in the X and Y directions. The fabric consists of cloth overlaid with coatings as per Figure 9.

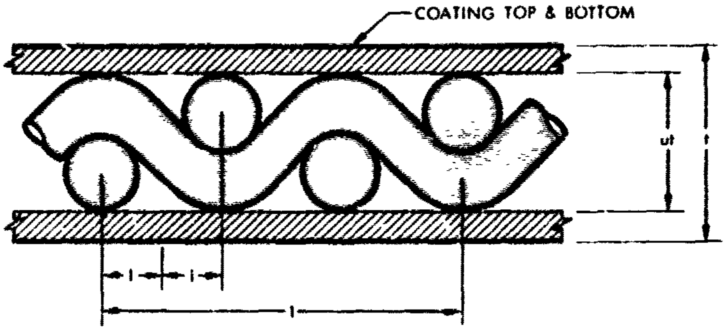

Figure 9. Idealized Cross-Section of Fabric

The length ℓ is not only a function of the distribution of fibers, but also a function of the internal pressure (ℓ increases with internal pressure).

The total area of a one-inch wide strip of fabric is $1 \times t$. The area of a fiber of circular cross section is $A = \pi D^2/4$, and it is assumed that $D_x = D_y = D$. The number of fibers per square inch is n^*. The number of fibers in a unit width is

$$n = n^* \frac{1 \times t}{1} = n^* t$$

This number, n, corresponds to a concentration, η, which is assumed to be known for the fabric.

$$\eta = \frac{\text{net area of fibers}}{\text{total area}}$$

$$= \frac{\frac{D^2 \pi}{4} \cdot n}{1 \times t} . \qquad (2)$$

Therefore,

$$n = \frac{4 \eta t}{\pi D^2} \text{ per inch of width} \qquad (3)$$

However, the material is coated on each side which reduces the actual thickness, t, to ut, where $u = 0.60$.

Therefore:

$$2D = ut = 0.60\,t$$
$$D = 0.30\,t$$

$$\eta_{new} = \eta \frac{1t}{1 \times ut} = 1.67\eta$$

The horizontal distance L between the center line of the fibers will be determined. The shortest distance is:

$$L = D = 0.30t$$

This condition corresponds to number of fibers being

$$n_D = \frac{1}{0.30t} > n = \frac{4\eta t}{\pi D^2} .$$

Therefore, the actual distance is

$$L = L_D \frac{n_D}{n} = 0.071 \frac{t}{\eta}$$

or if expressed in general in terms of u (rather than u = 0.6)

$$L = \frac{2\pi D^3}{4\eta u t^2} \approx 0.2 \frac{u^2 t}{\eta}$$

and consequently

$$\ell = \frac{L}{2} = 0.1 \frac{u^2 t}{\eta} \tag{4}$$

If $k \neq 1$, that means more layers exist and

$$\ell_k = 0.1 \frac{u^2 t}{k \eta} . \tag{4a}$$

Having ℓ_k, q_x and q_y, the following modification also applies:

$$P_x = \frac{q_x}{k},\ P_y = \frac{q_y}{k}$$

Using these equations the critical loads can be predicted. If, as a result of testing, the critical stresses

$$\sigma_I \text{ and } \tau_I$$

are known (σ = tensional stress, τ - shear stress), it can be determined with help of interaction curves, whether or not a given combination of actual stress σ and τ is satisfactory. The corresponding loadings P and V may be considered rather than σ and τ.

A typical interaction curve is given on Figure 5. Assume that the interaction curve is circular:

$$\left(\frac{P}{P_I}\right)^2 + \left(\frac{V}{V_I}\right)^2 = 1.$$

Then

$$P_{cr} = a\ P_I$$

$$a \quad \sqrt{1 - \left(\frac{V}{V_I}\right)^2} \tag{5}$$

This leads to

$$V = V_I \sqrt{1 - a^2} \tag{6}$$

TORSION

Consider an internally pressurized cylinder which is made of fabric as per Figure 10.

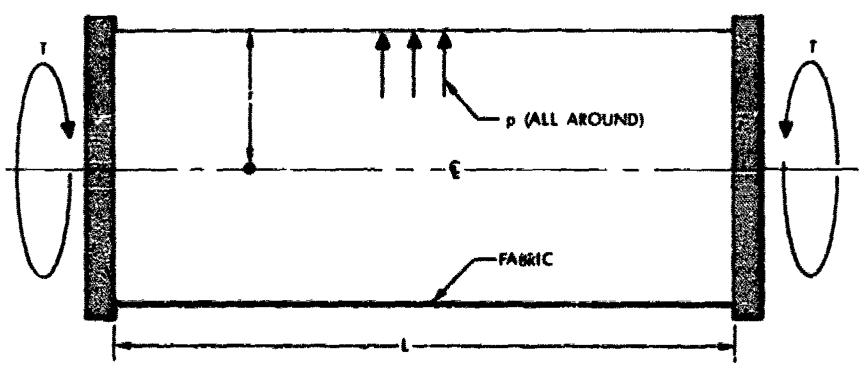

Figure 10. Torsional Test Element

A rigid bulkhead is uniformly assumed to be at each end of the cylinder to transmit torsional moment into the skin. Since the cylinder is pressurized, the skin will be stressed in the longitudinal and the circumferential directions.

$$\left.\sigma_{\ell} = pr/2t \qquad \sigma_c = pr/t\right\} ; \tag{7}$$

where

σ_{ℓ} = Tension stress in longitudinal direction (psi)

σ_c = Tension stress in circumferential direction (psi)

p = Internal pressure (psi)

r = Radius which corresponds to pressure p

t = Thickness of the fabric skin.

The thickness, t, is difficult to measure for the fabric; therefore, it is customary to work with the running loads

$$q_{\ell} = \sigma_{\ell} t, \; q_c = \sigma_c t \quad (\text{lbs/in.}) \tag{7a}$$

instead of with σ_{ℓ} and σ_c.

A torsional moment T (in lbs) is applied at the ends of the cylinder. This introduces a shear stress in the skin, in addition to the original q_{ℓ} and q_c, which act in the principal directions (directions of fibers)

$$q_{\tau} = \tau t \quad (\text{lbs/in.})$$

where

τ = Shear stress in psi

Figure 11 shows a small element, stressed with q_{ℓ}, q_c and q_{τ}, which is in equilibrium.

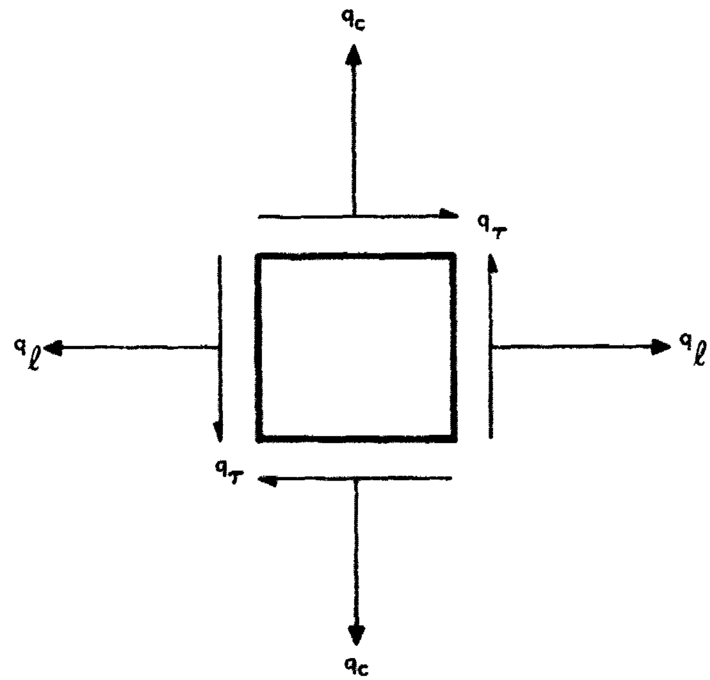

Figure 11. Differential Element Under Combined Stresses

The maximum amount of shear which can be added can be determined by using Mohr's circle. Consequently, the maximum permissible torsional moment T which will cause the first wrinkle in the skin can be found (See Figure 12).

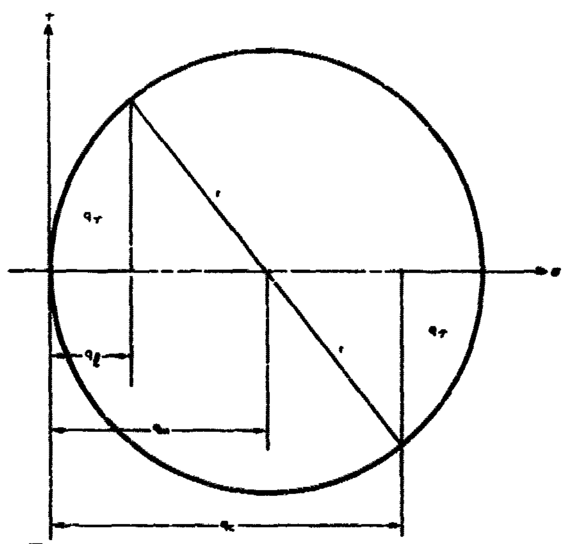

Figure 12. State of Stresses on Element

Since

$$q_m = 0.750 \text{ pr.}$$

$$q_\tau = 0.707 \text{ pr.} \tag{8}$$

The same result can be derived from the "combined action" formula:

$$q_\ell \, q_c = q_\tau^{\,2}$$

$$\left(\frac{pr}{2}\right) pr = q_\tau^{\,2}$$

$$q_\tau = 0.707 \text{ pr.}$$

On the other hand it is noted that

$$q_\tau = \frac{T}{2\pi r^2} \; ; \; T = 2\pi r^2 q_\tau .$$

By using equation (8), an expression can be developed for the maximum torsional moment which an inflated cylinder can take if the inflation pressure is p:

$$\max T = 4.45 \, pr^3 \tag{9}$$

The following general formula may be used for the corresponding rotation of the section

$$\theta = \frac{TL}{JG^*} \tag{10}$$

Where $G^* = Gt$

$J = 2\pi r^3$ (in.^{3})

G = Shear modulus (lbs/in.^{2})

L = Length of cylindrical element (in.)

t = Thickness of fabric (in.)

To justify the formulas derived above a cylinder 123.75 inches clear span between rigid bulkheads was tested.

The cylinder was pressurized with an internal pressure of 2 psi. Then, the pressure was increased to 5 psi; and torsional moment, T, in increments was applied. The corresponding rotations were measured. After reaching the critical value of torsion, torsion was removed and internal pressure was increased to 10 psi. Then, the cylinder was loaded in a similar manner. The cylinder was tested twice in torsion. Table 3 shows the sequence of loading. All deformations and corresponding diameters were measured. The cylinder withstood the predicted critical torsional moments, and no wrinkling was noted because of considerable local rigidity of the cylinder walls.

Table 3. Torsional Test Results

p psi	Diameter (inches) Before T was applied	Diameter (inches) After T was applied	Applied Torsional Moment (inch-lbs)	Predicted Values r inches	Predicted Values T_{wr} in.-lbs
2	14.6				
5	14.7	15.6	7500	7.52	9450
2	15.1				
10	15.3	15.3	7500	7.63	19800
2	15.2				
15	15.5	15.5	7500	7.80	31700
5	15.8	15.2	9450		
10	15.3	15.3	19800		
2	14.7				
15	15.4	15.4	19800		

The following phenomena were noted:

1. The decrease in diameter of the cylinder was negligible under torsional moment.

2. Under constant torsional moment, the rotation of the element continued for some time. This indicates that, for each load, a "lower" and "upper" value of deformation can be prescribed. The "lower" deformation is that which occurs immediately when the corresponding moment is applied. The "upper" occurs after the load is held constant and the inflatable member reaches equilibrium. This indicates that it is justifiable to introduce time-variable behavior.

3. After removal of the torsional moment, the angle of deformation does not completely disappear. There still is considerable permanent set remaining. The formula given previously for shear modulus is derived by averaging the angles of rotations. Table 4 shows the agreement between predicted values and actual test values:

Table 4. Comparison of Predicted and Test Values for G

p	Average G From Test (psi)		Predicted Value of G (psi)
	1st Loading	2nd Loading	
5	5550	4900	5430
10	8850	8000	7950
15	9350	9100	9350

BENDING

Assume that a long circular inflated cylinder with constant cross-section is simply supported at the ends as per Figure 13. The end bulkheads are assumed to be rigid in comparison with the skin.

After application of a bending moment, M, at the ends of the inflated cylinder deflections will be introduced into an originally straight member. The deflection increases with the applied moment until the first wrinkle in the skin occurs. The moment which theoretically causes the first wrinkle in the skin concludes the so-called prebuckling stage of deformation and is

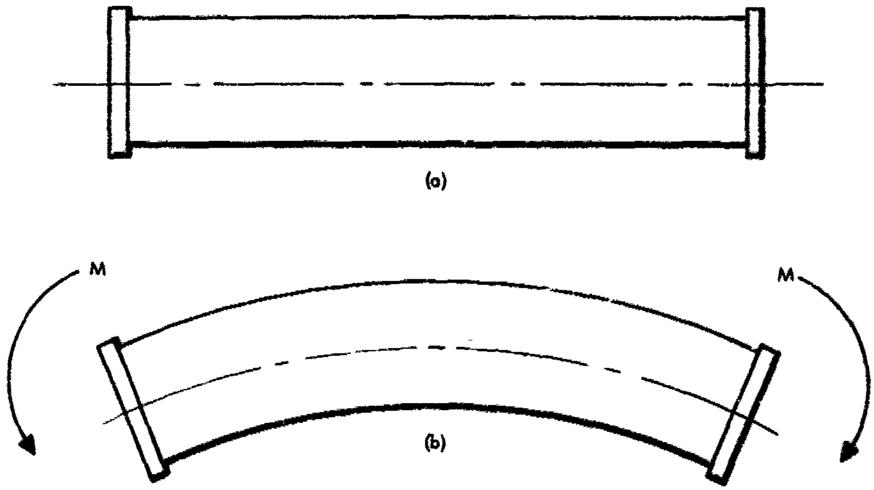

Figure 13. Bending Element

indicated with M_{wr}. This moment, however, usually does not cause failure of the element which can take a considerably larger moment, possibly up to $2M_{wr}$. It is safe, however, to use M_{wr} as a limiting design moment. Due to the internal pressure the skin is already prestressed before the application of the moment. The moment increases the stress by the value

$$\sigma_M = \pm \frac{M \cdot y}{I}$$

Using the value $q_m = \sigma_M t$ instead of σ_M and the value $I^* = I/t$ instead of I the above formula is rewritten as

$$q_M = \pm \frac{M \cdot y}{I^*} \quad \text{where } I^* = \pi r^3$$

The internal pressure does not contribute any restoring moment but simply stabilizes the membrane walls. All fibers remain in tension in the prebuckling stage:

$$\begin{aligned} &\text{at top:} && q_T = q_\ell + q_M \\ &\text{at bottom:} && q_B = q_\ell - q_M \end{aligned}$$

The following condition will be explored to determine M_{wr}:

$$q_B = q_\ell - q_M = \frac{pr}{2} - \frac{M_{wr} \cdot r}{\pi r^3} = 0 .$$

This leads to

$$M_{wr} = \pi p r^3/2. \tag{11}$$

This is the moment which theoretically causes the first wrinkle. Actually, the skin has considerable local rigidity and will sustain some compression stress before wrinkling. The first visible wrinkle may occur at a moment $M \approx 1.6 M_{wr}$. The deflection may be calculated in the prebuckling stage according to usual bending theory because the tests showed that in the prebuckling stage the moment of inertia I remains approximately unchanged and the section stays approximately circular. Also, the change of internal pressure due to bending in the prebuckling stage is negligible.

Tests were performed to justify the above formulas. The cylindrical test element has the following characteristics:

Material: 2 ply, 4.8 oz., 25 mil thick neoprene

Modulus of elasticity: E is 1793 lbs/in. (or 69000 psi)

Approximate thickness: $t = 0.026$ inches

Fabrication radius: $r_o = 6.7$ inches (at 2 psi pressure)

Theoretical span: 148 inches

Bending tests were conducted at three different pressures: p = 5, 10, and 15 psi. The moment was applied at each end on metal bulkheads with an eccentric axial load, which was counterbalanced by an additional concentric axial load. In each case the element was loaded in bending until excessively large deformations occurred. At this point the test was stopped to avoid the possibility of damage to the test specimen. Except in the last case (15 psi), the applied moment exceeded the theoretical wrinkling moment. In the tabulated data the first value given is the theoretical wrinkling moment; the second value is the maximum moment applied in the test. In no case did an actual wrinkle occur; therefore, the inflated element is capable of taking at least the calculated moment M_{wr}. The deflections were in fair agreement with the calculated values.

Table 5. Bending Test Results

p (psi)	r (inches)	$M_{wr} = \frac{\pi p r^3}{2}$ and M_{actual}	$I = \pi r^3 t$	Deflections (inches) Calculated	Deflections (inches) Test Results
5	7.52	3,350	34.7	3.82	
		5,100		5.83	5.93
10	7.63	7,000	36.4	7.64	
		8,496		9.20	8.53
15	7.80	11,250	38.7	11.50	
		10,200		10.45	10.46

The permanent set was not considerable; the "upper" and "lower" deflections were noted, but were not as important as in the case of torsion.

For discussion purposes consider an inflated fabric cylinder with metallic bulkheads. The cylinder is assumed to be loaded concentrically with an axial compressive load P. The initial imperfection at the end is indicated by e. The modulus of elasticity E_ℓ only will be considered because E_c probably will not have a significant influence on the results.

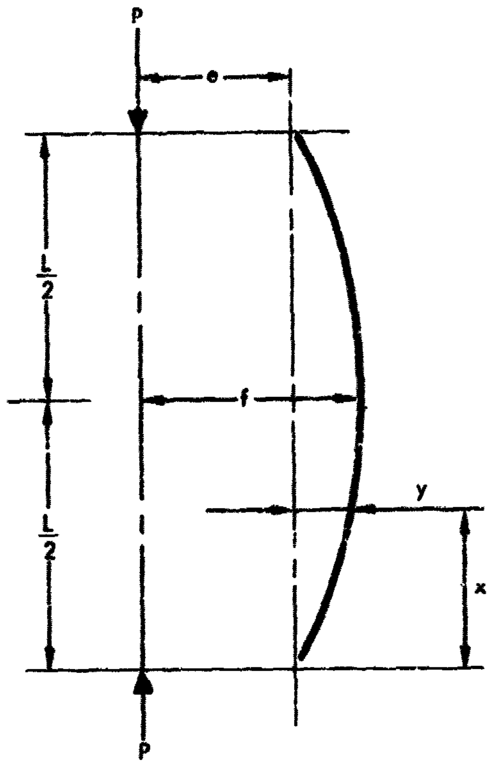

Figure 14. Column Under Axial Compression

Start from the well known differential equation (Figure 14):

$$EI \frac{d^2y}{dx^2} = -P(e + y) .$$

and consider that zero stress in the skin occurs when:

$$\frac{M}{\pi r^2 t} + \frac{P}{2\pi rt} = \frac{pr}{2t} .$$

This indicates that the moment which the skin can carry is:

$$M = \frac{\pi r^2}{2} \left(pr - \frac{P}{\pi r} \right)$$

or

$$M_{wr} = \frac{r}{2} (\pi r^2 p - P_{act}) \tag{12}$$

Solution of the differential equation leads to the well-known equation of Euler:

$$P_E = E \frac{\pi^3 r^3 t}{L^2} \tag{13}$$

This means that as long as the wall is supported by internal pressure, it acts as a thin-walled tubular column and the Euler equation defines the critical buckling load. The internal pressure required to stabilize the wall and make it capable of taking the load P is defined by the following equation:

$$\sigma_l = \frac{pr}{2t} - \frac{P}{2\pi rt} \geq 0$$

This leads to "limiting load" P_L which corresponds to condition, $\sigma_l = 0$:

$$P_L = \pi r^2 p \,. \tag{14}$$

The limiting load does not depend on L but on p and r, only. It should be noted that when using formulas (13) and (14) the actual value of the radius, r, should be used, including the increase in radius due to pressurization of the cylindrical element. Therefore, for certain types of fabric material, the pressure versus elongation curve must be known in advance.

The load, P_E, is independent of the internal pressure but limited by P_L. This relation can be shown by a diagram as in Figure 15 which shows a plot of L versus P.

Also, the relation of p versus P may be shown with diagram as per Figure 16.

Other relationships which exist in the theory of stability regarding the fixity of supports probably are valid for inflated columns, too. Therefore, P_E should be multiplied by the factor, C, which depends on the end-fixities. For example:

C = 0.25 for column with one end fixed, the other free

C = 1.00 for both ends hinged

C = 4.00 for both ends fixed

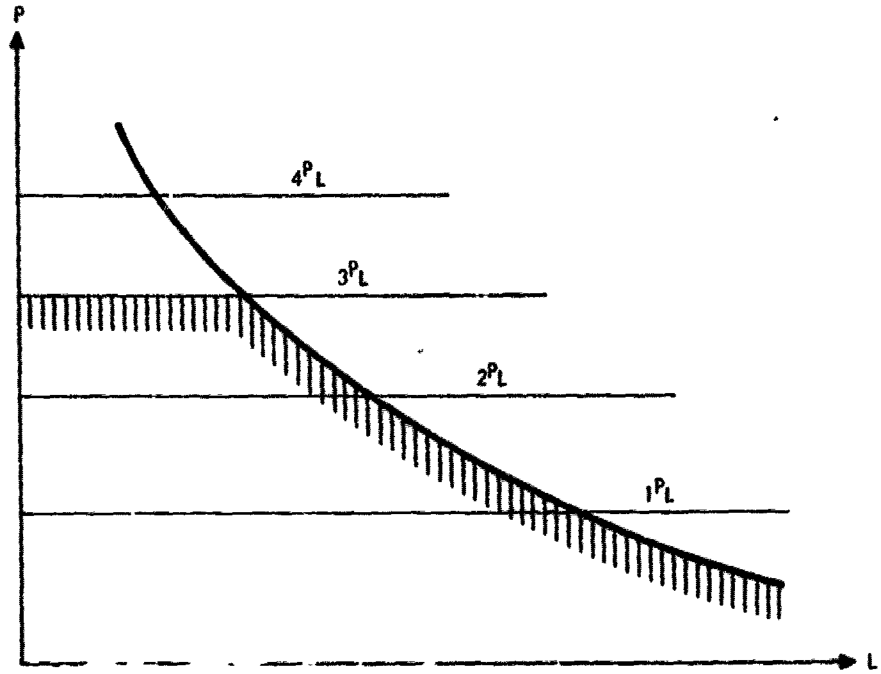

Figure 15. P Versus L Diagram

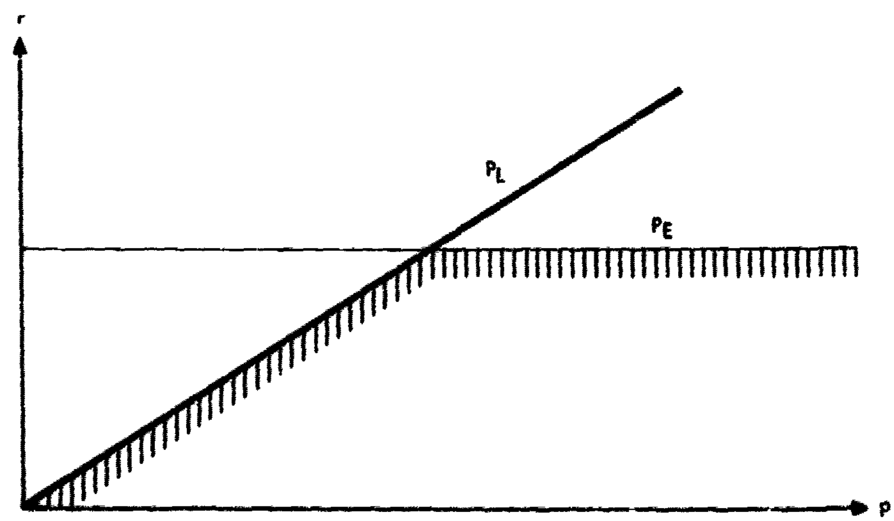

Figure 16. P Versus p Diagram

To determine the critical load on an inflated fabric column consider the following formulas:

$$CP_E = CE_\ell \pi^3 r^3 t/L^2 \text{ and } P_L = \pi r^2 p$$

If $CP_E < P_L$ assume $P_{cr} = CP_E$

If $CP_E > P_L$ assume $P_{cr} = P_L$.

For verification of above derivations a test was conducted on a fabric cylinder with rigid bulkheads as was described previously. The ends were pinned. The test was performed at three different pressures. The following schedule gives the comparison between predicted critical loads and the test results.

Table 6. Column Test Results

p (psi)	r (inches)	P_E (lbs)	P_L (lbs)	P_{actual} (lbs)		Note:
5	7.52	1080	885	630	600	Computed P_{cr} is underlined
10	7.63	1125	1825	1117	909	
15	7.80	1200	2860		1180	Testing was performed twice

Agreement is satisfactory with the maximum discrepancy noted at the smaller pressures.

It was shown that the internal pressure does not have any influence on the Euler load other than to stabilize the walls against local buckling up to a certain limit. This was considered by introducing the limiting load P_L. This leads to the hypothesis that in general the "beam-column" theory may be applicable to inflated members, too, with slight modifications. Therefore, the general differential equation for an elastic line may be used but with limitation:

$$EI \frac{d^2y}{dx^2} = -M \leq M_{wr} \tag{15}$$

where

$$M_{wr} = \frac{r}{2} (\pi r^2 p - P_{act})$$

To justify this equation a test was performed at three pressures on the same element which was described previously. The load was applied as per Figure 17.

An axial load of 400 lbs was selected for all three pressures. The moment was applied in increments until M_{wr} was reached. Table 7 presented below shows the predicted and actual results for the deflections.

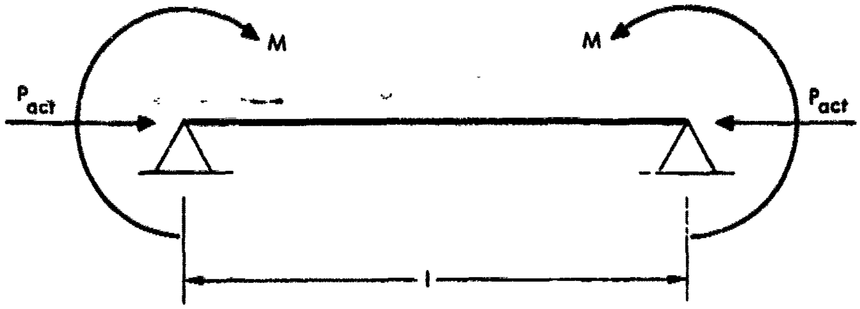

Figure 17. Beam Column

Table 7. Beam Column Test Results

p (psi)	M_{wr} (inch-lbs)	Deflection Predicted (inches)	Deflection Measured (inches)	
			Lower	Upper
5	1840	2.20		3.8
10	5310	5.42	5.9	6.8
15	9600	9.30	9.20	9.7

It is noted that there is a considerable difference between the "lower" and "upper" deflections. Predicted values are in good agreement with the measured "lower" values. At 5 psi the "lower" deflection was not measured because this fact was not realized at this time. Since M_{wr} values were exceeded and no wrinkle occurred, the predicted values of the moments can be considered to be on safe side.

It was also noted in the axial test that when the Euler load was reached, immediate collapse occurred. It was also noted in the bending test that the first wrinkle did not occur at M_{wr}, therefore M_{wr} hardly can be called the "critical moment," because the inflated beam probably can take much more moment. The present case (axial load with bending) can be regarded as an intermediate case. The beam can take more moment than M_{wr}, but not as much as in the case of pure bending.

EFFECT OF DEGREE OF FIXITY OF SUPPORTS ON COLUMNS

Usually a structure does not consist of just one beam, but a combination of beams. Some of the beams may be recognized as "primary," some as "secondary," built into the "primary" system. These structures are

more complicated than simple beams, and additional problems are introduced, such as:

1) Interaction between inflatable beams of the system under external loading. This requires an interaction analysis.

2) Influence of fixities of supports for each beam on deflections and stresses.

3) Design of junctions.

4) Behavior of inflated cylinders in the postbuckling stage.

This study is of primary importance because if the design of a complicated system of beams is based on the appearance somewhere of the first wrinkle, the weakest member will limit the whole system and make it uneconomical. If, on the other hand, a wrinkle can be allowed to occur in a secondary member, the economy of the whole system will be increased. This introduces a very complex set of problems about the behavior of inflatable cylinders in the postbuckling stage.

The scope of this publication does not permit treatment of these problems, but some will be mentioned. The most important problem is the effect of end fixity of the columns.

In practice the ideal hinge or ideal fixed support does not occur. Usually, the actual condition at the support is somewhere in between these ideal cases. The critical load can be expressed as:

$$P_{cr} = CP_E . \tag{16}$$

When the load P reaches the critical load P_{cr}, the support will oppose any rotation of the ends that may develop and keep the column in straight shape. No resistance to bending is given by simple supports (hinges) and for this case the column will bend without end resistance.

The ideally fixed support provides resistance to bending. The reactive moment developed at the ends of the column which oppose bending can be expressed as

$$M = k\theta \tag{17}$$

where M = Moment

k = Coefficient of proportionality

θ = Inclination angle.

The coefficient of proportionality, k, is a spring constant of the support in in. lbs/rad. For simple supported ends $k = 0$ and $C = 1$. For ideally fixed ends $k = \infty$ and $C = 4$. For elastically fixed supports, k usually can be calculated or estimated. It is possible to derive the relation between k and C, which leads directly to a determination of P_{cr} for elastically supported columns. Assume that the column is loaded with an axial load $P = P_{cr}$ and is in a deflected equilibrium shape. It is assumed that a reacting moment is applied at the ends of the column and tends to return the deflected column to previous straight equilibrium configuration. This moment is indicated by $M = k\theta$ as shown in Figure 18.

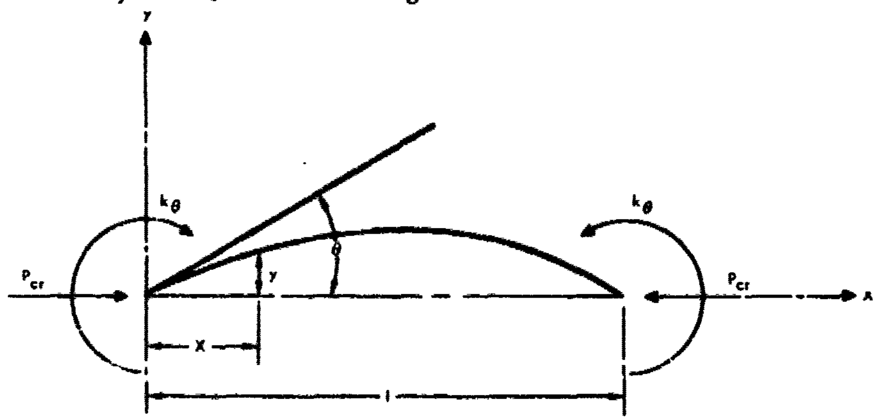

Figure 18. Geometry for Fixity Calculations

For any value x, the expressions for M_x and deflection y can be written:

$$M_x = -Py + k\theta = EI\,\frac{d^2y}{dx^2}$$

$$\frac{d^2y}{dx^2} + \frac{P}{EI}\,y = \frac{k}{EI}\,\theta$$

Solution of this differential equation leads to a mathematical expression for k:

$$k = -EI\sqrt{\frac{P_{cr}}{EI}}\cdot \mathrm{Cot}\left(\frac{l}{2}\sqrt{\frac{P_{cr}}{EI}}\right), \tag{18}$$

and finally the relation between k and C is:

$$\frac{k}{P_E l} = \frac{-1}{\pi}\sqrt{C}\cdot \mathrm{Cot}\left(\frac{\pi}{2}\sqrt{C}\right). \tag{19}$$

This relation can be shown graphically as in Figure 19 in which C is given as a function of k/P_E.

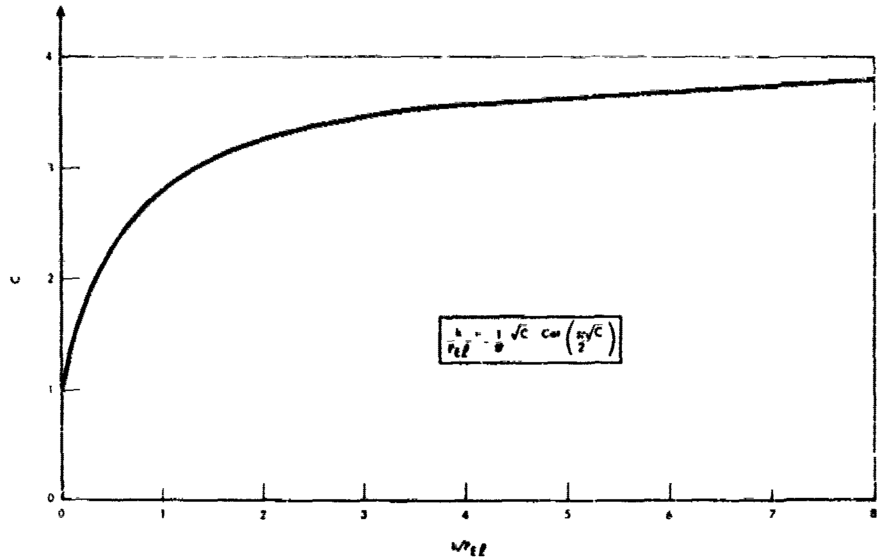

Figure 19. Determination of C-Factor

This permits the finding of $P_{cr} = CP_E$ for any elastically fixed column, provided the degree of fixity on each support is the same.

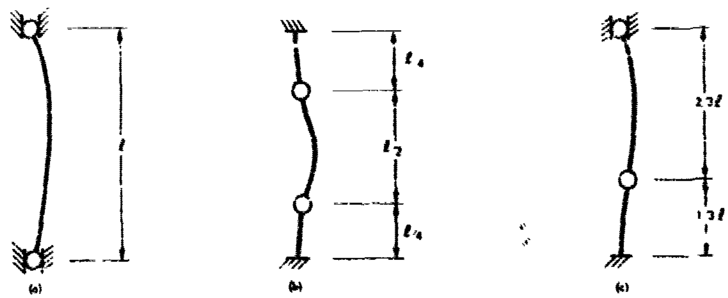

Figure 20. Various Fixity Cases

Referring to Figure 20, it is known that for case (a) the critical load will be the Euler load. It is also known that for case (b) the critical load will be the Euler load for the same column, except that ℓ is replaced by $\frac{\ell}{2}$, the distance between inflection points which are indicated by circles on Figure 20.

$$P_{cr} = 4\,\frac{\pi^2 EI}{\ell^2}$$

Similarly, the column shown under (c) is equivalent to the column with pinned ends of the length 0. 71 ℓ

$$P_{cr} = 2.04 \frac{\pi^2 EI}{\ell^2} .$$

Therefore, any column with know end fixities permits the determination of the length between inflection points, which can be designated with ℓ'. The critical load for the column is:

$$P_{cr} = \frac{\pi^2 EI}{(\ell')^2} .$$

From Figure 20 it is evident that:

If to fixity as per (a) corresponds $\ell_1' = \frac{\ell}{1} = \frac{\ell}{n_1}$

and to fixity as per (b) corresponds $\ell_2' = \frac{\ell}{2} = \frac{\ell}{n_2}$

then to "mixed" fixity, as per (c) will correspond $\ell_3' \approx \frac{1}{1.43} = \frac{1}{n_3}$

where

$$n_3 = \frac{n_1 + n_2}{2.1} .$$

But

$$\frac{\pi^2 EI}{(\ell')^2} = C \frac{\pi^2 EI}{\ell^2}; \; (\ell')^2 = \frac{\ell^2}{C}; \; \ell' = \frac{\ell}{\sqrt{C}} = \frac{\ell}{n}$$

Consequently

$$n = \sqrt{C} .$$

Therefore, if the column is supported so that each support has a different fixity, then for each fixity (1) and (2) there is a corresponding C_1 and C_2. Then:

$$n = \frac{n_1 + n_2}{2.1} \text{ or } \sqrt{C} = \frac{\sqrt{C_1} + \sqrt{C_2}}{2.1}$$

$$C_{effective} = 0.227 \left(C_1 + C_2 + 2\sqrt{C_1 C_2}\right). \tag{20}$$

For example for the column as per (c)

$$C_{eff.} \approx 0.227\,(1 + 4 + 2\sqrt{1 \times 4}) = 2.04$$

This approximate formula permits analysis of columns with different fixities which may be very helpful for a designer. For the extreme case represented by $C_1 = C_2 = 4$, $C = 3.6 < 4$ is obtained; for another extreme case represented by $C_1 = C_2 = 1$, $C = 0.91 < 1$ is obtained. In both extreme cases the coefficient obtained is 10% conservative, but this is the maximum possible deviation.

CONCLUDING REMARKS

In this paper a preliminary and approximate procedure was presented for analysis of inflatable cylinders under various loadings, which should be of some help to the design engineer. It is hoped that the procedure can be improved when more test results will be available.

Study of the Memory Effect of Polyethylene for Space Expandable Structure Applications*

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ABSTRACT

The memory effect of irradiated polymers involves the ability of a formed object to return to its original shape merely by heating, after it has been heated above ambient, deformed and cooled in the deformed shape. This novel phenomenon has high potential utility in space expandable applications. The parameters controlling the minimum restoration temperature, degree of restoration and restoring force have been studied with consideration of the technique for expansion of objects in space. Polyethylene has been studied although the process is applicable to many polymers. The minimum restoration temperature is controlled by the inherent polymer composition and nature. The degree of restoration is controlled by the restoration temperature and by the deformation conditions. The restoring force is controlled by the deformation conditions and the radiation dose.

INTRODUCTION

Project ECHO has involved the launching into space of a compactly folded, aluminum-coated plastic which, after being sent aloft, opens into a sphere which is then employed as a passive communications satellite. The folded, balloon-like structure, is induced to open by the sublimation of an organic solid material under the vacuum conditions of high altitude (ca. 1000 miles). The projected diameter of ECHO I, after expansion, was 100 feet, and of ECHO II is 135 feet. The former employed a mixture of benzoic acid and anthraquinone as the sublimating solids, while the latter, to be launched later in 1963, will employ acetamide (ref. 1). ECHO I was composed of 0.5 mil thick mylar covered by a layer of aluminum 2 x 10^{-4} cm. thick, ECHO II is composed of 0.35 mil thick mylar sandwiched between two 0.18 mil thick layers of aluminum foil. The balloon is carefully folded on

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Earth and placed into a 36 inch long magnesium alloy container which is caused to open at a predetermined height. The polymeric balloon is released and the organic chemicals, dispersed throughout the folds as a powder, volatilize upon being subjected to the sudden vacuum. Release of the gases causes the balloon to expand.

One of the many problems associated with such a venture involves the heavy payload required to be sent aloft (ECHO I weighed 135 lbs. and ECHO II will weigh 500 lbs.). In addition, the required sublimating solid adds to the total launch weight. One method of weight savings has involved the etching of holes in the aluminum coating. This produces a grid-like structure, rather than a continuous one, but does not affect the satellite's ability to serve as a radio wave reflector. Another problem, which developed as a result of the ECHO I experiment, involved the wrinkling of the skin of the satellite, probably by the pressure of solar energy. The external forces causing this wrinkling of the exposed portion of the polymer are due primarily to radiation pressure and stagnation pressure (ref. 2). The wrinkling problem in ECHO II was minimized by employing the aluminized laminated structure that avoided any exposure of polymer. Although wrinkling was prevented, the weight was increased considerably.

In theory, an ideal approach to solving the weight and solar pressure problems (for any object to be expanded in space), is one where the polymer could expand without the aid of any sublimating solids, and additionally, where the polymer itself could be grid-like in nature. The weight requirements would, therefore, be at an absolute minimum. This approach means that the entire satellite would be noncontinuous in nature.

Such an accomplishment would be impossible with the present approach. However, one way does exist to bring this about. It involves the well known, but little studied, memory effect of polymers.

MEMORY EFFECT

The memory effect has been referred to briefly in the literature with reference to polyethylene (refs. 3-5). When polyethylene is irradiated (crosslinked) and heated to a temperature above its normal melting point (T_m), it will no longer melt. It can, however, be deformed, and if cooled in the deformed state, will maintain the new shape. A second heat treatment will cause the polyethylene to return to its original state. The return of the polyethylene to is original shape, after being deformed, is called the "memory" effect.

It is apparent that the memory effect, if it could be applied to a passive communications satellite, would solve the problems due to heavy payload and solar pressure. The polymer could be prepared as a grid-like structure, coated or laminated with the thin aluminum layer, heated, and deformed by folding compactly on Earth. It would then be sent aloft and released into the vacuum of space. Expansion in space would be akin to the occurrence of memory. Heat for expansion to occur would be obtained from the sun. It is further apparent that this approach would allow the expansion of any shaped object, not necessarily spherical ones.

Very little data on the memory effect of polymers is available and direct application of the phenomenon to this problem has not been possible. A fundamental study has been required. Such a study was sponsored by NASA. Among the questions to be answered from this study were the following:

(a) What is the minimum restoration temperature (MRT) that is required for memory to occur?

(b) What parameters control the degree of restoration of the polymer, and what is their relative importance?

(c) What is the magnitude of the restoring force, and can it be controlled?

(d) Is radiation necessary for the memory effect, or can memory be induced in polymers without the necessity of crosslinking?

It is with this purpose in mind that the fundamental study of the memory effect of polyethylene was undertaken.

PARAMETERS FOR STUDY

The memory effect was studied by (a) subjecting standard sized specimens to radiation, (b) heating them in an oil bath, (c) deforming by folding in half, (d) cooling, and (e) allowing them to restore (i.e., exhibit their memory). Three major sets of information were sought: (1) the minimum temperature for complete restoration (MRT), (2) the degree of restoration, and (3) the restoring force as a function of various parameters.

These three functions were studied by controlling the five parameters listed above:

(a) Radiation dose was varied; unirradiated specimens were also tested.

(b) Conditions of heating prior to actually deforming were

varied. These were called the deformation conditions, and involved the temperature of the deformation bath and the time interval the specimen was in the bath.

(c) The type of deformation to be discussed here was performed only by folding in half and therefore is constant.

(d) Cooling conditions (slow or rapid) after the performance of the deformation were varied.

(e) Restoration was varied in the same manner as the deformation conditions; by varying the temperature in the bath and the time interval involved.

In actuality, for simplification, the time interval prior to deformation and during restoration was kept constant; only the deformation temperature and restoration temperature were of importance.

EXPERIMENTAL METHODS

The minimum restoration temperature was studied by varying the restoration temperature (step e) and visually observing when complete restoration occurred.

The degree of restoration was studied by means of an angle measurement. After steps (a), (b) and (c) were performed, step (d) was allowed to take place with the folded specimen being placed next to a protractor (Figure 1). The percent restoration in terms of an angle could be ascertained.

The restoring force was measured by determining the ability of the deformed specimen to move a mass. After steps (a), (b) and (c) were performed, lead masses were placed on the polymer and its ability to restore (step d) with the mass on it was measured (Figure 2). In practice, the maximum mass moved and the minimum mass not moved were both determined (to within 10% of each other).

The large majority of experiments was performed with a single polyethylene grade (Grex 50-050), and the effect of all the above parameters was obtained employing this polymer. Other polyethylene grades were employed to determine the effect of polymer molecular weight and density on the MRT, degree of restoration, and restoring force.

RESULTS AND DISCUSSION

This study of the parameters that control the polyethylene memory phenomena indicated the following:

1. <u>Minimum Restoration Temperature</u>: This temperature,

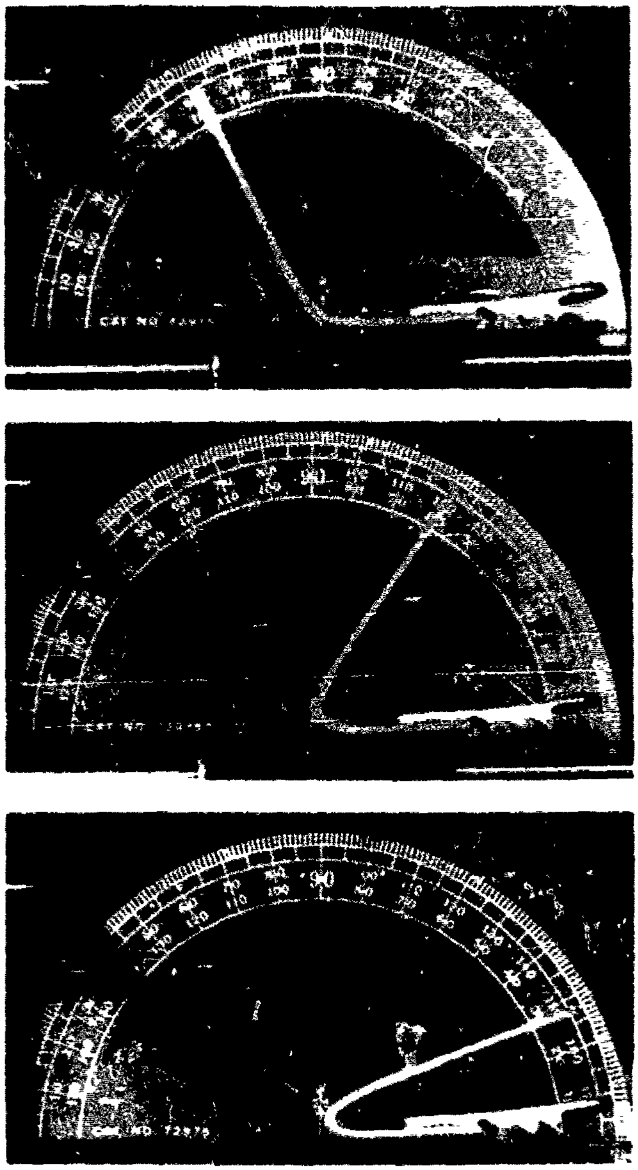

Figure 1 DEGREE OF RESTORATION PROCEDURE FOR STUDYING MEMORY EFFECT

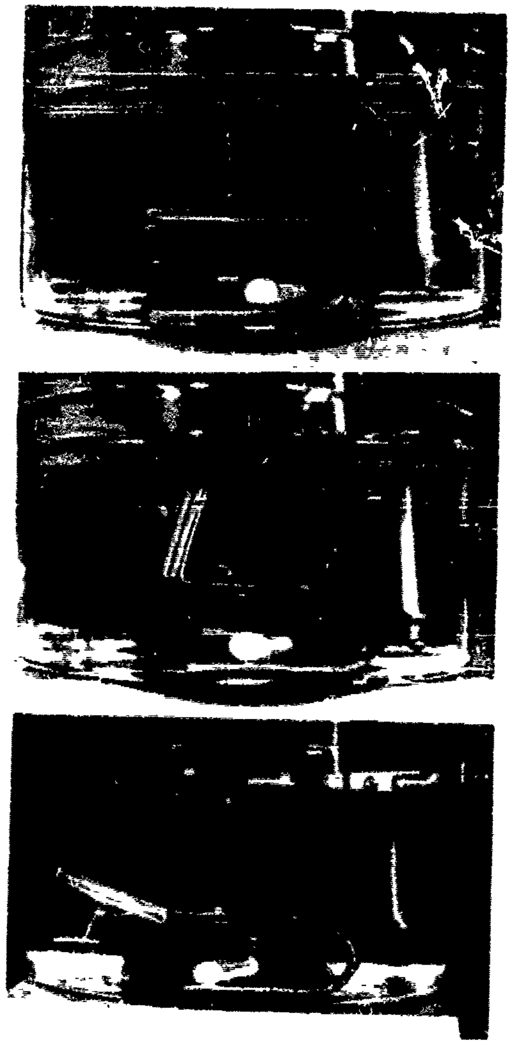

Figure 2 RESTORING FORCE FOR STUDYING MEMORY EFFECT

for irradiated polyethylene, is the normal melting temperature (T_m) of the unirradiated polymer. Since the T_m of unirradiated low density polyethylene is ca. 105°C. and of unirradiated high density polyethylene is ca. 125-130°C., these are the minimum temperatures required for complete restoration. Aside from the requirement of a minimum radiation dose of 5-10 Mrads, the MRT is independent of all other parameters.

Crosslinked polyethylene copolymers with ethyl acrylate or vinyl acetate, which exhibit lower T_m values, exhibit lower MRT values. Crosslinked polypropylene exhibits its MRT at ca. 180°C. Therefore, to apply the memory effect to expansion in space applications, the restoration temperature required can be controlled by proper choice of polymer composition. Radiation treatment to induce crosslinking is all that would be further required to impart the ability to restore (memory).

2. Degree of Restoration: The degree of restoration is controlled primarily by the restoration temperature. As indicated above, this value must be at least as high as the normal T_m of the polymer. Under these conditions, the radiation dose, deformation temperature and quenching conditions after deformation are not factors in the quantitative degree of memory.

The restoration behavior of polyethylene is quite complex if restoration temperatures below the normal crystalline melting point are employed. The degree of restoration is, then, not complete, but decreases with decreasing restoration temperature (see Figure 3). However, the overall behavior is intimately related to the deformation temperature. At any specific restoration temperature, lower deformation temperature causes a greater degree of restoration.

The exact reasons for this behavior are beyond the scope of this article, and discussed elsewhere (ref. 6). However, it can be pointed out that when deformation is performed below T_m, the overall quantity of melted and recrystallized regions present is less than if the T_m had been attained. This means that the ability to prevent restoration is reduced, since unmelted crystalline regions do not act in this manner. This further implies that the lower the deformation temperature, the greater the degree of memory, which is the observed phenomenon.

A further observation was that the onset of restoration occurs more rapidly as the deformation temperature is lowered.

Applying this knowledge to expansion in space, it is

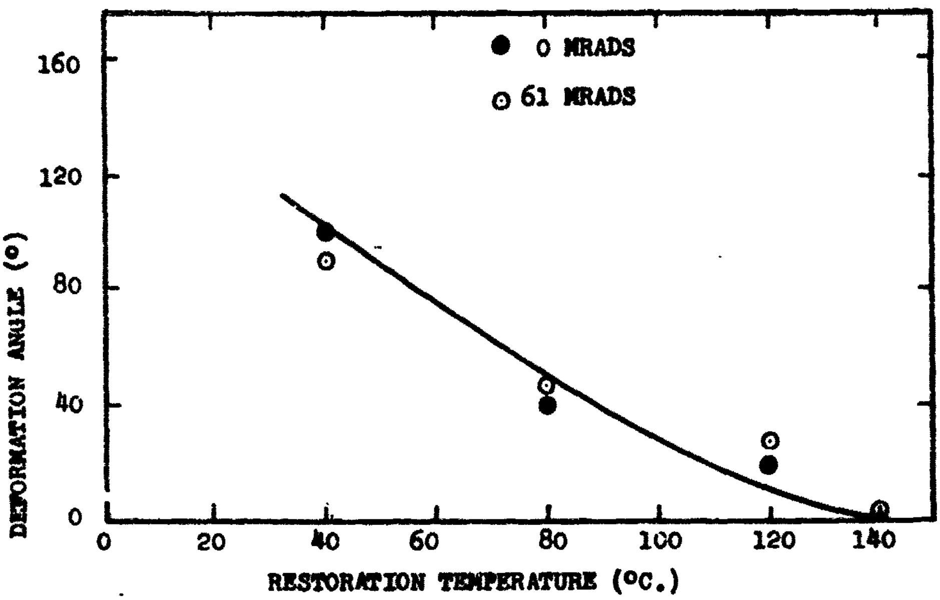

Figure 3 RESTORATION TEMPERATURE EFFECT ON THE DEGREE OF RESTORATION

evident that a variety of potential approaches become available. The onset of memory and the total extent of memory that occurs, can be determined by controlling various parameters on earth prior to launching into orbit. These parameters are the polymer nature, radiation dose and deformation conditions.

3. Restoring Force: The restoring force is controlled primarily by the deformation temperature, specimen thickness and the radiation dose. No other factors are of significance.

The restoring force increases with decreasing deformation temperature (see Figure 4). The reasons for this are the same as those that explain the deformation temperature effect on the degree of memory. The restoring force also increases with increasing thickness (see Figure 5) and radiation dose (see Figure 6). As a semi-quantitative estimate of the forces involved, 1 cm. x 7 cm. x 0.127 cm. specimen, folded as described above, exhibits a restoring moment of ca. 300 gm.-cm. These forces, in this thickness range, (0.05-0.20 cm.) increase linearly with the square of the thickness (Figure 5). The restoring forces also increase with increasing polymer density and molecular weight.

CONCLUSIONS

From the above discussion, it is apparent that the polyethylene memory effect is, indeed, a complex phenomenon. The minimum restoration temperature is a function of the polymer nature. The parameter that controls the degree of restoration is the restoration temperature, although a minimum radiation dose is required. The parameters that control the restoring force are primarily the deformation conditions, thickness and radiation dose. Complete memory can be brought about only by heating to a temperature equal to or greater than the point at which the polymer ceases to remain crystalline. Maximum restoring force can be induced by employing low deformation temperatures, high radiation dose and thicker films. However, limitations exist in space expandable applications to all these aspects. If the deformation temperature is too low, memory will not be locked in and restoration will occur too rapidly; too high a radiation dose induces brittleness in the polymer; too thick a polymer brings about weight problems, which this approach designed to eliminate. A proper balance of the requirements must be obtained. It is apparent, however, that the technique, as shown by this exploratory study, is applicable for space expandable structures. It is also important to note that different polymers crosslink with different proficiencies when subjected to irradiation. Polyethylene crosslinks rather well, but polyisobutylene

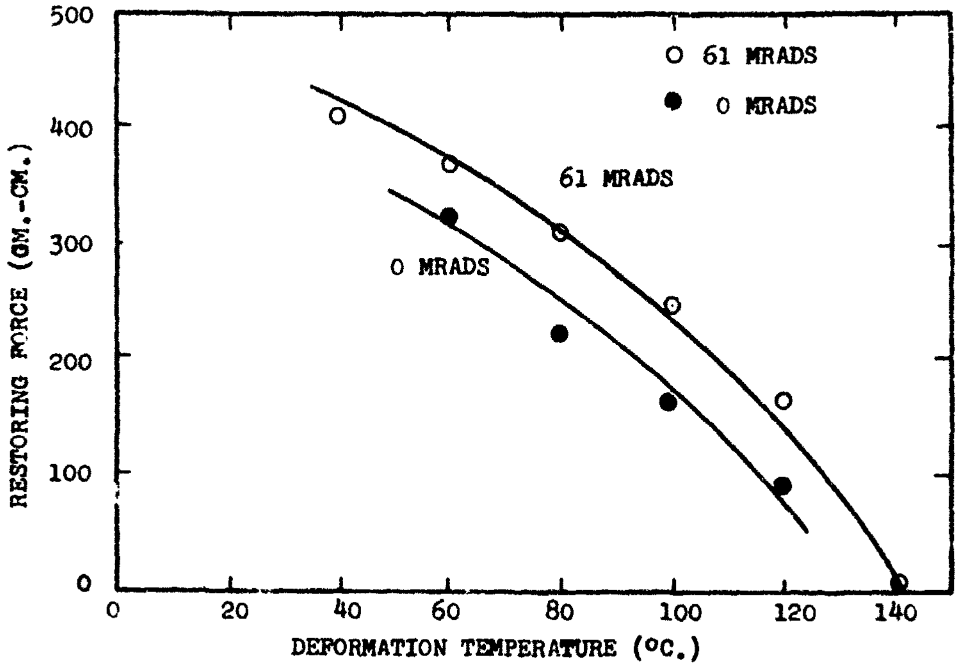

Figure 4 DEFORMATION TEMPERATURE EFFECT ON THE RESTORING FORCE

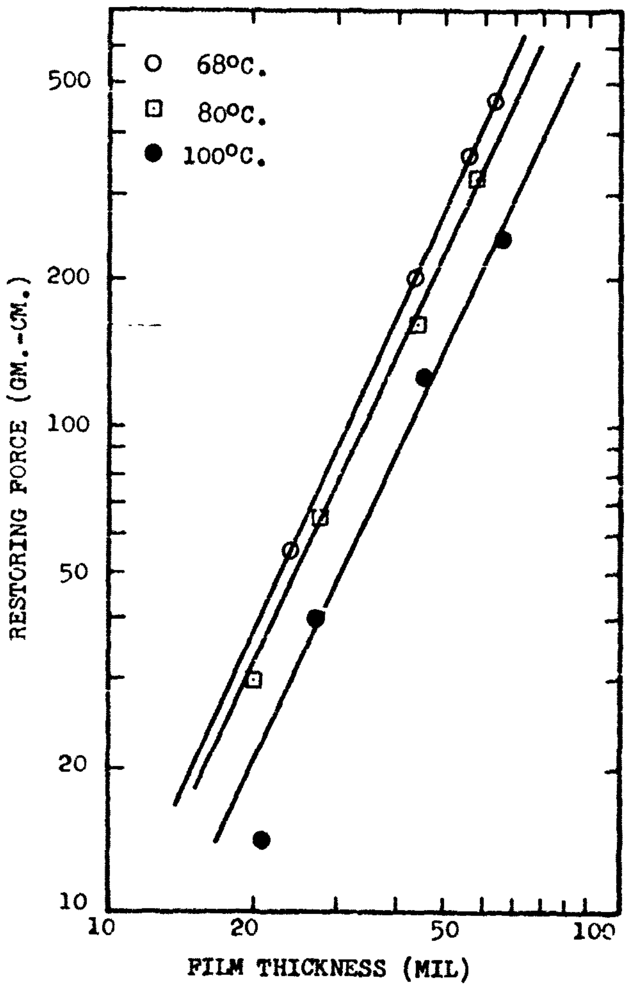

Figure 5 FILM THICKNESS EFFECT ON THE RESTORING FORCE

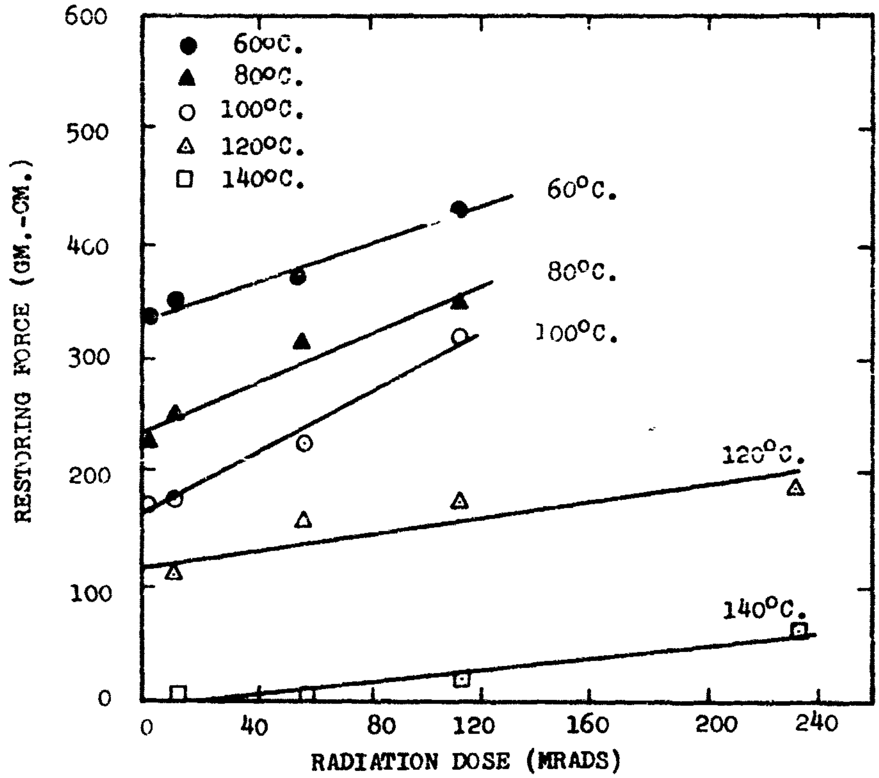

Figure 6 RADIATION DOSE EFFECT ON THE RESTORING FORCE

cr polymethyl ... crosslink. Th ... certain additi ve. ... the normal ten ... be reversed. hi. ... ng approach has been studied in detail by P I (ref. 7). Thus, applicability ... memory effect to a wide variety of polymers is ...

As a result of our study, a novel approach ... obtaining spa expandable structures, of any s..., has been investig ed. A new method for obtaining weight savings upon aunching is possible. Further insight into the memory ef ect of polymers, with particular reference ... space expanda le applications, has been brought abou .

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